MSC-PA-R-69-2
Supplement 8

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 9 MISSION REPORT

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

(NASA-TM-X-70270) APOLLO 9 MISSION REPORT. SUPPLEMENT 8: DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION (NASA) 65 p

N74-75209

Unclas
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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
August 1970

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DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

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MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
August 1970

11176-H266-RO-00

PROJECT TECHNICAL REPORT

APOLLO 9

LM-3

DESCENT PROPULSION SYSTEM FINAL FLIGHT EVALUATION

NAS 9-8166

16 June 1969

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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PREFACE

This report is published as Supplement 8 to the Apollo 9 Mission Report (MSC-PA-R-69-2).

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PURPOSE AND SCOPE

The purpose of this report is to present the results of the postflight analysis of the Descent Propulsion System (DPS) performance during the Apollo 9 Mission. The primary objective of the analysis was to determine the steady-state performance of the DPS under the environmental conditions of actual space flight.

This report documents additional analysis of the DPS. Preliminary findings were reported in Reference 1. This report also brings together information from other reports and memorandums analyzing specific anomalies and performance in order to present a comprehensive description of the DPS operation during Apollo 9.

The following items are the major additions to, or changes from, the results as reported in Reference 1:

1) The performance values for the first DPS burn are revised.
2) The analysis techniques, problems and assumptions are discussed.
3) The flight analysis results are compared to the preflight performance prediction.
4) The Propellant Quantity Gaging System performance is discussed in greater detail.
5) The start and shutdown transient performance for DPS operation are included.

SUMMARY

The performance of the LM-3 Descent Propulsion System during the Apollo 9 Mission was evaluated and found to be satisfactory.

The drop in the regulator outlet pressure experienced during the early portion of the first burn has been attributed to a blockage in the internal heat exchanger of the supercritical helium tank. This blockage occurred during prelaunch servicing. The blockage cleared approximately 35 seconds after ignition. The operation of the pressurization system during the remainder of the burn was considered satisfactory.

The engine thrust chamber pressure oscillations experienced during the second engine burn has been attributed to helium ingested in the engine propellant feed lines. This occurred due to either lateral or rotational maneuvering of the LM after separation from the CSM while the zero gravity propellant retention screens were not covered, thus allowing helium to be ingested into the engine. The chamber pressure stabilized prior to the end of the burn and the third burn was normal.

A helium leak developed in the supercritical helium system after launch. This leak was attributed to a line brazing failure which occurred due to the shock caused by the activation of the isolation squib valve.

The steady-state performance was determined by analyzing a segment of the FTP portion of operation during the first burn using the Apollo Propulsion Analysis Program. Thrust, specific impulse and mixture ratio were within the predicted three sigma limits.

The engine performance corrected to standard inlet conditions for the FTP portion of the first burn was as follows: thrust, 9809 pounds, specific impulse, 303.3 seconds, and propellant mixture ratio, 1.593. These values are 0.31, 0.10, and 0.0 percent different, respectively, from the

values reported from acceptance tests of the engine and were within specification limits.

The Propellant Quantity Gaging System appeared to be within the predicted accuracy with the exception of the Oxidizer Tank No. 2 probe. This gage appeared to have a bias of at least 1.2%.

Based on the results of this analysis and recent ground tests, it is concluded that for performance predictions, the degradation of specific impulse with respect to throat erosion at FTP should be increased.

INTRODUCTION

The Apollo 9 Mission was the ninth in a series of flights using specification Apollo hardware. It was the second flight test and the first manned flight of the Lunar Module (LM). The mission was the third manned flight of Block II Command and Service modules (CSM) and the second manned flight using a Saturn V launch vehicle. The overall objectives of the mission were to evaluate crew operations of the Lunar Module and to demonstrate docked vehicle functions in an earth orbital mission, thereby qualifying the combined spacecraft for lunar flight. Combined spacecraft functions included Command Module docking with the Lunar Module, spacecraft ejection from the launch vehicle, five Service Propulsion System (SPS) firings while docked, a docked Descent Propulsion System (DPS) firing, and extravehicular crew operation from both the lunar and command modules. Lunar module operations included a complete rendezvous and docking profile and an Ascent Propulsion System (APS) firing to propellant depletion.

The space vehicle was launched from the Kennedy Space Center (KSC) at 11:00:00 a.m. (EST) on March 3, 1969. Following a normal launch phase, the S-IVB stage inserted the spacecraft into an orbit of 102.3 by 103.9 nautical miles. The CSM docked with the LM and the docked spacecrafts were ejected from the S-IVB approximately four hours after launch. During the next 25 hours, four SPS burns were performed. Approximately 50 hours after launch, the first DPS maneuver, the docked burn, was performed. The burn duration was 369.7 seconds and resulted in a velocity change of 1741.7 ft/sec. The docked burn included long fixed throttle position (FTP) operation and manual throttling. The SPS was again fired approximately five hours later. At 92:39:36 ground elapse time (GET) the spacecrafts were separated in preparation for the rendezvous maneuvers. At approximately

94 hours after launch, the DPS performed a phasing maneuver burn, 19.7 seconds in duration and accomplishing a velocity change of 89.3 ft/sec. This burn began at the minimum throttle setting and ended at approximately 40% throttle. Two hours later the DPS performed an insertion maneuver burn of 22.3 seconds at the minimum throttling setting. The velocity change was 42.2 ft/sec. The insertion maneuver ended the DPS mission duty cycle. The descent stage was separated from the ascent stage a short time later. The APS performed two firings, the latter being to propellant depletion and the SPS performed three more burns during the subsequent portion of the mission.

The actual ignition and shutdown times for the three DPS firings are shown in Table 1. The throttling profiles for the first and second DPS maneuvers are shown in Figures 1 and 2.

The Apollo 9 Mission utilized LM-3 which was equipped with DPS Engine S/N 1030. The engine and feed system characteristics are presented in Table 2.

Each DPS burn was preceded with a two jet +X LM Reaction Control System (RCS) ullage maneuver to settle propellants.

There were two Apollo 9 Mission Detailed Test Objectives (DTO) specifically related to the DPS.

M11.6 LM PGNCS/DAP Performance and Thrust Performance.

The functional test objectives of this DTO were:

1) Verify the ability of the Digital Auto Pilot (DAP) to control rates and altitudes during docked and undocked DPS firings.
2) Evaluate the accuracy of thrust vector control (TCV)

using the DPS with the PGNCS DAP gimbal trimming capability active.

3) Verify that DPS thrust transient response to manual throttle commands is satisfactory for the lunar landing mission.

M13.12 DPS Burn Duration Effects and Primary Propulsion/Vehicle Interactions.

The functional test objectives of this DTO were:

1) Demonstrate that a long duration burn at the fixed throttle point (FTP) can be satisfactorily accomplished in the space environment.

2) Verify that engine performance prediction techniques are adequate for lunar mission performance predictions.

3) Verify the DPS propellant feed, pressurization and propellant gaging system for the lunar landing mission.

4) Verify that no adverse propulsion/vehicle interactions exist in space environment.

The detailed requirements of this objective are described in Reference 2.

STEADY-STATE PERFORMANCE

Analysis Technique

The major analysis effort for this report was concentrated on determining the flight steady-state performance of the DPS during the fixed throttle position (FTP) mode of operation of the first burn (Docked). The throttled portions of the first burn, the second (Phasing) burn and third (Insertion) burn were of insufficient duration at a given throttle position to allow for a meaningful detailed performance analysis. The performance analysis was accomplished by use of the Apollo Propulsion Analysis Program which utilizes a minimum variance technique to "best" correlate the available flight and ground test data. The program embodies error models for the various flight and ground test data that are used as inputs, and by iterative methods arrives at estimations of the system performance history and propellant weights which "best" (minimum-variance sense) reconciles the available data.

Analysis Program Results

The Apollo Propulsion Analysis Program results presented in this report are based on simulations using data from the flight measurements listed in Table 3. Comparison of the inflight chamber pressure to the engine acceptance test and computed values indicates that the flight transducer may have incurred a zero shift and drift error probably due to thermal effects during the burn. Chamber pressure transducer errors (as great as 2%) due to thermal effects have been noted in ground tests although not to the degree seen on the LM-3 flight. Because of this apparent error, the flight chamber pressure could not be used in the performance analysis. Figure 3 presents the measured chamber pressure during the initial and throttling portions of the docked burn. The propellant densities were calculated from sample specific gravity data from KSC, flight bulk propellant temperatures of 69°F, assumed interface temperatures of 69°F and 65.5°F for the oxidizer and fuel respectively, and the flight interface pressures. The measured interface temperatures were questionable in absolute magnitude since the bulk oxidizer temperature did not agree with the interface oxidizer temperature. It was, therefore, assumed that the interface temperatures were biased but that the difference in temperature between the oxidizer interface and the fuel interface was correct. A difference between the two is expected due to the helium/fuel heat exchanger. The preliminary estimated spacecraft damp weight (CSM and LM minus the DPS propellants) at ignition of the docked burn was obtained from the Apollo Spacecraft Program Office. The weight of the CSM was reduced by approximately 365 lbs based on performance analysis of the third SPS maneuver (Reference 5). The damp weight was held constant throughout the burn. The initial estimates of the DPS propellants on board at the beginning of the analyzed time segment was extrapolated (based on a simulation of the first portion of the burn) from the loaded propellant weights.

The DPS steady-state performance was determined from the analysis of a 160-second segment of FTP portion of the burn. The segment of the burn analyzed commenced approximately 55 seconds after DPS ignition (FS-1) and included the flight time between 49:42:30 (178950 sec) and 49:45:10 (179110 sec) G.E.T. The first 55 seconds of the burn were neglected to reduce the transient nature of the data resulting from both engine throttling and a supercritical helium pressurization anomaly which occurred in the early portion of the burn. The latter portion of the FTP operation was not included due to a 0.5 psi step increase in the oxidizer interface pressure.

This pressure increase could not be explained and was not substantiated by other pressure measurements. Further, the computed acceleration trend resulting from this pressure increase did not agree with the measured acceleration trend.

The results of the Propulsion Analysis Program simulation of the docked DPS burn are presented in Table 4 along with the preflight predicted values. The values presented are end point conditions of the segment analyzed and are considered representative of the actual flight values throughout the segment. As can be seen in Table 4, the actual values are, in general, less than predicted. There are three reasons for this. 1) The regulator outlet pressure used in the predicted performance (supplied by GAEC) was approximately 4 psi higher than measured in flight. It has been determined that the inflight value agrees well with tests performed by the regulator manufacturer and that calibration errors at the GAEC test facility were responsible for the erroneous value used in the prediction.
2) Recent tests have indicated that the error model describing the degradation of Isp with respect to engine throat erosion was somewhat optimistic. The model was updated during the postflight analysis to reflect a greater drop in Isp due to throat erosion. 3) Based on poor acceleration and pressure trend matches during early postflight simulations, it was determined that the actual engine throat erosion was greater than predicted. The throat erosion rate was thus increased to best match the overall trends of the available pressure measurements. At the end of the segment analyzed, the throat erosion was 1.1% greater than predicted. For the total burn time (FS-1 + 215 seconds), this difference is well within the 3σ predicted uncertainty value of 2.1%. Figure 4 presents a comparison of the predicted throat erosion and the inflight estimated erosion.

The engine performance during the burn was satisfactory and was as expected in view of the decreased interface pressures and increased throat erosion.

Critique of Analysis Results

Figures 5 through 16 show the analysis program output plots which present the filtered flight data and the residual errors, or differences between the filtered data and the program calculated values. The figures presented are thrust acceleration, oxidizer interface pressure, fuel interface pressure, oxidizer delta pressure (from tank bottom to interface), fuel delta pressure, oxidizer injector inlet pressure, fuel injector inlet pressure, quantity gaging system for oxidizer tanks 1 and 2, and the quantity gaging system for fuel tanks 1 and 2. The filtered chamber pressure data is also presented without residual errors since this measurement was not used in the analysis. It is presented to show a magnified view of the transducer error observed in flight.

A strong indication of the validity of the analysis program simulation can be obtained by comparing the thrust acceleration as determined from the LM Guidance Computer (LGC) ΔV data to that computed in the simulation. Figure 5 shows the thrust acceleration derived from the ΔV data and the residual error between the measured and computed values. The time history of the residual error has an essentially zero mean and a small, but acceptable, positive trend.

Several problems were encountered with flight data while analyzing the steady-state performance at FTP. Several assumptions were necessary in order to obtain an acceptable match to the flight data. These problems are discussed below.

Pressures and temperatures measured in the frequency modulated (FM) and pulse amplitude modulated (PAM) modes, when digitized, were subject to unusual bias and scale effects. Based on a review of the oscillograph traces, it was concluded that these problems were caused by data reduction. The source of these problems has not been determined.

The oxidizer interface temperature (PAM) appeared to be low by approximately 2.7 degrees when compared to the oxidizer bulk temperature (PCM). Both temperatures were nearly constant during the entire burn. The fuel interface temperature was approximately 4 degrees less than the oxidizer interface temperature. Whereas a difference between fuel bulk and fuel interface temperature is expected due to the helium/fuel heat exchanger, a difference in oxidizer interface and oxidizer bulk temperature is not. It was thus decided to bias both the interface temperatures by approximately 2.7 degrees.

The delta-pressure measurements (from tank bottom to interface) appeared low when compared to expected values considering the inflight interface pressures. Their trend with time, however, agreed well with the interface pressure trend. Based on the interface pressures and the measured acceleration data, the delta-p's were biased up by 1.54 and 0.85 psia for oxidizer and fuel respectively.

The injector inlet pressure appeared high when compared to expected values. A bias downward of approximately 5 psia was required to adjust the magnitude of the measurements to agree with the interface and gaging system (i.e. flowrate) measurements. The trends of the injector inlet measurements did not agree with the trends of the interface pressure or the delta-p measurements. It is felt that this particular disagreement may be due to data reduction problems rather than actual flight measurement disagreement. This was somewhat substantiated by a comparison of the chamber pressure as recorded in the PCM mode to that recorded in the FM mode where a slight trend difference could be seen. Efforts to resolve the FM and PAM data reduction problems will be continued.

The performing of the long DPS burn in the docked configuration severely compromised the capability of the Performance Analysis Program to distinguish between spacecraft mass and engine thrust. This is because the rate of change of vehicle acceleration is critically small especially when compared to the normal mode of operation (undocked). Thus, in the analysis, it was necessary to hold the estimate of initial spacecraft weight fixed.

Comparison With Preflight Performance Predictions

Prior to the Apollo 9 Mission the expected inflight performance of the DPS was presented in Reference 4. The preflight performance report was intended to bring together all the information relating to the entire Descent Propulsion System and simulate its operation in the space environment.

The predicted steady-state and related three sigma dispersions for the specific impulse, mixture ratio and thrust during the FTP portion of the docked burn is presented in Figure 17. The corresponding analysis program computed flight performance values are also presented for comparison. Both the thrust and the specific impulse as derived from flight data are less than predicted. This is because of the erroneously high regulator outlet pressure used in the preflight simulation and the effect of greater throat erosion than predicted. The mixture ratio is reasonably close to the predicted value as expected since neither throat erosion or regulator outlet pressure significantly effect mixture ratio. All values are within the three sigma predicted limits.

Engine Performance at Standard Inlet Conditions

The flight performance prediction of the DPS engine was based on the data obtained from the engine acceptance tests. In order to provide a common basis for comparing engine performance, the acceptance test performance is adjusted to standard inlet conditions. This allows actual engine performance variations to be separated from performance variations which are induced by the pressurization system and propellant temperature variations. The standard inlet conditions performance values were calculated for the following conditions.

Standard Inlet Conditions

Oxidizer interface pressure, psia	221.7
Fuel interface pressure, psia	221.5
Oxidizer interface temperature, °F	70.0
Fuel interface temperature, °F	70.0
Oxidizer density, lbm/ft^3	90.24
Fuel density, lbm/ft^3	56.45
Thrust acceleration, lbf/lbm	1.0
Throat area, in^2	54.4

The following table presents ground test data and flight test data adjusted to standard inlet conditions. The differences between the two sets of performance values as derived from ground test data are because of the following. 1) The difference in throat area assumed. When correcting to standard inlet conditions, the engine manufacturer bases the computation on the pre-acceptance test throat area which varies from engine to engine. The values computed using the DPS characterization were based on the often quoted reference area of 54.4 square inches. In general, this difference is small. 2) Density differences. There is a small difference in the

	GROUND TEST		FLIGHT
	Test Facility[1] Data Reduction Program	Engine Prediction Characterization	Analysis Results
Thrust, lbf	9736	9779	9809
Specific Impulse, sec.	303.0	303.0	303.3
Mixture Ratio	1.596	1.593	1.593
Thrust Coefficient, C_f		1.780	1.796
Ox. Characteristic Velocity, C*, ft/sec		5476	5433

specific gravities of the propellants used in the engine acceptance tests and the Apollo 9 Mission. 3) Engine hydraulics resistance differences. During the acceptance tests, at the time slices selected for data reduction, the engine hydraulic resistances are computed from the measured values of interface pressure, chamber pressure, propellant flowrates and temperature. These resistances are then used in correcting the data to standard inlet conditions. The computed values of these resistances will change from time slice to time slice. The values presented in the above table from Reference 4 are the average of two acceptance test data slices and are meant to be representative of the engine. The values of the resistances used in the DPS characterization are the result of the summation of all the data from the acceptance test, thus resulting in a single value for each of the two resistances which best characterizes the engine. These values should be used to compute the standard against which the flight results can be compared.

Since the characterization data is the best estimate of engine characteristics prior to the flight, the differences between it and the flight

[1]As reported in Reference 4.

data should be within the engine's repeatability, the uncertainty associated with the characterization, and the uncertainty associated with the flight analysis results.

Comparing the engine flight performance at FTP during the Docked Burn corrected to standard inlet conditions against the values as given by the DPS characterization shows the flight data to be 0.31% greater, 0.10% greater, and 0.0% for thrust, specific impulse and mixture ratio, respectively. These differences are due to the 0.90% increase in C_f and 0.79% decrease in C* derived from the flight analysis. These differences are well within the engine repeatability uncertainties as well as the performance specifications.

The difference in thrust may well be due to a flight analysis assumption with regard to spacecraft weight. Due to the low rate of change of vehicle acceleration, the estimate of the initial spacecraft weight was not permitted to vary in the analysis, thus permitting the possibility of some weight error.

Since the Apollo 9 flight was the first opportunity to make a detailed evaluation of the DPS inflight performance, and since the results of the analysis were within the expected dispersions of the preflight analysis, it is concluded that the only change to the presently used performance prediction techniques should be to increase the degradation of specific impulse with throat erosion at FTP. This conclusion will be reviewed, however, as ground test and flight test data becomes available.

SECOND BURN HELIUM INGESTION[1]

During the second DPS maneuver (phasing burn) engine roughness was experienced for approximately 2.5 seconds as the engine was being throttled from 12 to 40 percent thrust. The cause of the roughness has been attributed to engine helium ingestion. The measured chamber pressure for this burn is presented in Figure 18. The second DPS burn was initiated at 93:47:34.7 (337654.7 sec) GET at the minimum throttle setting (approximately 12.7%). Approximately 2.5 seconds after ignition, the throttle setting was manually increased towards 40%. At 93:47:39.22 (337659.22 sec) GET, a bubble of gaseous helium and propellant vapor approximately 130 cubic inches in size began to flow through the oxidizer flow control valve. It has been determined that at engine ignition, this gas bubble was located approximately four feet upstream of the engine oxidizer interface. The fluid flowing through the oxidizer flow control valve changed from a liquid in cavitation to compressible gas flow allowing an increased volumetric flowrate through the valve. This greatly increased the propellant flowrate, both upstream and downstream, of the flow control valve. In response to the increased flow, the oxidizer injector inlet pressure increased from 120 to 215 psia and the chamber pressure increased from approximately 30 to 40 psia (Figure 19). At the same time, the oxidizer pressure drop between tank bottom and engine interface was noted to increase while the oxidizer interface pressure decreased. It was the increased chamber pressure at 93:47:39.70 (337659.70 sec) GET that gave the first noticable indication of thrust chamber pressure oscillations. In response to the chamber pressure increase, the fuel injector inlet pressure increased from approximately 39 to 54 psia. At 93:47:39.70 GET, a second bubble of gaseous helium and propellant vapor

[1]References 3, 6 and 7.

approximately 65 cubic inches in size reached the fuel flow control valve. At engine ignition, this bubble was located at the helium/fuel heat exchanger. Similar in effect to the oxidizer flow, the fuel injector inlet pressure increased from approximately 54 to 160 psia, the chamber pressure increased from 40 to 45 psia, and the fuel pressure drop from tank bottom to engine interface increased while the fuel interface pressure decreased.

The chamber pressure began to decrease at 93:47:39.9 (337659.9 sec) GET as the gas bubble began to flow into the combustion chamber. The oxidizer and fuel injector inlet pressures reacted due to the low chamber pressure of approximately 10 psia by decreasing 35 psi and 20 psi below normal values respectively. By 93:47:40.25 (337660.25 sec) GET, all helium had passed the flow control valves and the engine again reached steady state conditions at 93:47:41.7 (337661.7 sec) GET.

The third DPS maneuver (Insertion Burn) was performed without a recurrence of the helium ingestion observed during the phasing burn. All system pressures were smooth and nominal,indicating that the engine had not been degraded by pressure oscillations. The chamber pressure for the third burn is presented in Figure 20.

The most probable way in which the helium could have entered the feed lines was due to lateral or rotational accelerations on the spacecraft. Propellant flows from each tank through separate lines which meet at common points upstream of the engine interface. The length of these propellant tank connecting lines is approximately 9.5 feet. Zero gravity propellant retention devices are located inside the tank and cover the feed line inlets. These zero-g cans have been designed to retain propellant under negative accelerations equal to at least one earth g if the acceleration vector is parallel to the tanks. However, if the acceleration is lateral (perpendicular) to the tanks and if gas surrounds the retention devices, they are good to only 0.001 g for oxidizer and 0.003 g for fuel. In the undocked mode, a lateral movement of the spacecraft using two RCS thrusters would generate accelerations in excess of 0.003 g due to both engine thrust and acceleration head effects on the 9.5 feet connecting lines. The total acceleration has been determined to be as high as 0.01 g. Thus, if a pocket of helium surrounds the zero g cans, propellant in the connecting lines could be displaced by gas. This condition was highly likely since each tank had an ullage volume of approximately 60% and prior activities had tended to settle the propellants toward the top of the tanks.

The conditions that occurred during the LM-3 flight should not occur during the nominal lunar mission since prior to the long braking burn, the ullage volumes will be small.

PRESSURIZATION SYSTEM EVALUATION[1]

The performance of the supercritical helium pressurization system was considered satisfactory with the exception of the following:

1) The helium regulator outlet pressure and the engine interface pressures exhibited an unexpected decay during the initial portion of the first descent engine maneuver (docked burn).

2) An average pressure decay of approximately 2.9 psi/hr in the supercritical helium tank was indicated during the coast period between the first and second DPS firings. Normally, it is expected that the pressure should rise due to heat leak into the tank.

The ambient start bottle was loaded with approximately 1.1 lbm of helium at a pressure of 1591 psia at approximately 70° F. At launch, the pressure was approximately 1584 psia. Seven days prior to launch, the oxidizer and fuel tank pressures were increased from their load pressures to 160.1 and 169.6 psia respectively. At launch, the propellant tank pressures had decreased to approximately 144 and 162 psia, respectively. Two days prior to launch, the supercritical helium (SHe) tank was loaded with approximately 48 lbm of liquid helium at a pad pressure of about 87 psia. At launch, the pressure had risen to approximately 403 psia. The SHe tank pressure increase during this period was approximately 7.8 psi/hr due to normal heat leak into the system from the surrounding environment.

At 48 hours after launch, prior to pre-burn propellant tanks pressurization, the ambient helium bottle pressure was 1577 psia, the SHe tank pressure was 717 psia, the oxidizer tank pressure was 107 psia and the fuel tank pressure was 145 psia. The pressure decay in the ambient helium bottle was probably due to temperature changes, while the pressure decay in

[1]References 1, 3, 8, and 9.

the propellant tanks was attributed to helium going into solution. The average SHe tank pressure rise from launch was approximately 6.1 psi/hr. Upon activation of the ambient start bottle, the pressures increased to 234 and 235 psia in the oxidizer and fuel tanks respectively.

Internal Heat Exchanger Freezing

At engine ignition of the first DPS burn, the SHe tank pressure was approximately 743 psia. During the first 33 seconds of the burn, the SHe tank pressure decreased to approximately 711 psia while the regulator outlet and engine interface pressures decreased from 235 to 180 psia (Figure 21). If the helium system had operated correctly, the regulator outlet pressure would have been maintained at a constant value of 243 psia and the SHe tank pressure would have increased due to heat transfer through the internal heat exchanger. If no helium had flowed, the SHe tank pressure would have remained essentially constant at a pressure of 743 psia. Analysis indicates that during this initial portion of the burn, less than one pound of helium flowed from the tank. Temperature data indicates that this initial flow by-passed the internal heat exchanger.

Figure 22 presents SHe system temperatures monitored during the first portion of the DPS Docked Burn. The thermocouple located at the helium bottle outlet showed a drop after ignition indicating helium flow. The relatively low rate of change in the temperature indicated a low flowrate. The internal heat exchanger thermocouple was pegged at the maximum value of -65° F for the first 35 seconds of the burn. This was expected because the helium had made one pass through the external (helium/fuel) heat exchanger. The thermocouple located at the outlet of the internal heat exchanger showed the same trend as the measurement at the SHe tank outlet except at a higher temperature. If flow had occurred through the internal heat exchanger, the temperature at its outlet would have been appreciably lower than measured. No change in the helium temperature was recorded further downstream at the regulator inlet indicating that the second and final pass through the external heat exchanger had warmed the helium to the fuel temperature of approximately 69° F.

Approximately 32 seconds after engine ignition, a small rise in the temperature was seen at the regulator inlet followed by a rapid drop in temperature. Both the inlet and outlet temperatures of the internal heat exchanger experienced similar drops. This indicated a surge of relatively cold helium through the system. The small increase in temperature at the regulator inlet is probably due to compression of the helium at this point because of the initial pressure increase caused by the surge. The SHe tank pressure also began to rise a few seconds later. Following the initial surge, temperatures approached the anticipated operating levels of the system.

These events during the first burn have been determined to be the result of a blockage caused by partial freezing of the internal heat exchanger. The blockage was cleared approximately 35 seconds after ignition allowing the regulator outlet pressure to rise to the proper operating level. The blockage occurred during SHe tank servicing prior to launch when air was inadvertently drawn into the helium lines and froze in the SHe tank internal heat exchanger. Similar occurrences of internal heat exchanger blockage have occurred in ground tests. The ground support equipment and servicing procedures have been modified to eliminate the possibility of heat exchanger freeze-up in the future.

Supercritical Helium Tank Leak

Upon completion of the first DPS burn the SHe tank was isolated from the propellant tanks for the remainder of the mission by closing a solenoid valve just upstream of the pressure regulator.

The SHe tank pressure monitored during the coast period following the DPS Docked Burn indicated that the pressure decreased at an average rate of approximately 2.9 psi/hr. Since the SHe tank pressure should increase during a coast due to the absorption of heat from the engine and surrounding environment, a leak in the system was indicated.

Tests performed at WSTF indicated that the leak, determined to be approximately 0.1 lb/hr, must have been upstream of the internal heat exchanger. The leak could not have been an internal helium leak into the tank vacuum jacket since this situation would cause a tank pressure rise rather than a pressure decay. The most likely point at which the leak could have occurred was at the squib valve which isolates the SHe tank from the remainder of the system prior to use. It is postulated that the leak began when the squib valve was activated immediately following DPS engine ignition for the first burn. It is suspected that at least one occurrence of this nature has happened in ground tests when a failure of an internally brazed squib valve was found during drop tests of LM-2. The failure was a crack in the brazing material which was thin in the failed area. The time of the failure cannot be ascertained. However, it was most likely caused by the shock of the squib firing to pressurize the Descent Propulsion System. Except for the Apollo 5 mission (LM-1), which had no indication of a helium leak, the flight configuration of SHe tank, squib valve, bimetallic fitting and associated plumbing had not previously been tested together for the launch and boost vibrations, squib valve firing shock and thermal shock environment.

The Apollo 9 SHe squib valve, like the LM-2 squib valve, had valve fittings which were internally brazed preventing inspection of the joints.

Apollo 10 and subsequent squib valves are externally brazed. It is expected that this failure will not recur in view of the presently used techniques.

During the period between the first and second DPS maneuvers, the oxidizer interface pressure increased from approximately 242 to 257 psia while the fuel interface pressure increased from approximately 240 to 244 psia. These pressure increases are attributed to vaporization of propellants and thermal expansion of the gaseous helium in the tanks. Vapor pressures are approximately 14.8 and 2.2 psia at 70° F for the oxidizer and fuel respectively.

PQGS EVALUATION AND PROPELLANT LOADING

Propellant Quantity Gaging System

At engine ignition of the docked burn, all propellant gages were reading the maximum level of 95%. Shortly after the throttle was moved to the FTP setting, the gages began to indicate propellant consumption at approximately 29, 33, 29, and 27 seconds after ignition for Oxidizer Tank No. 1 (Ox 1), Oxidizer Tank No. 2 (Ox 2), Fuel Tank No. 1 (Fu 1), and Fuel Tank No. 2 (Fu 2), respectively.

At approximately 33 seconds after ignition the gages were reading 94.5, 95.0, 94.1 and 93.5% for Ox 1, Ox 2, Fu 1, and Fu 2, respectively. This indicated that initially there was approximately 0.5% more oxidizer in Ox 2 than in Ox 1 and 0.6% more fuel in Fu 1 than in Fu 2. The difference in the oxidizer tanks increased with time such that at the end of the burn, there was approximately 3.0% difference with the Ox 1 gage reading 40% and the Ox 2 gage reading 43%. Had the initially indicated unbalance between the tanks been real, the connecting balance line between the oxidizer tanks should have caused the quantities to converge with time. Engine thrust vector direction can cause propellant unbalance, but during the docked burn, the vehicle center of gravity and engine gimbal angles were such as to preclude this possibility. It was concluded, therefore, that the initial differences were due to system inaccuracies. A like conclusion was made for the indicated fuel tank unbalance since the quantities in Fu 1 and Fu 2 first converged and then diverged such that at engine shutdown the Fu 1 gage read 40.7% and the Fu 2 gage read 41.5%.

The approximate percent rates of change during the FTP portion were 0.182, 0.173, 0.183 and 0.174 %/sec for the Ox 1, Ox 2, Fu 1 and Fu 2 gages, respectively, and indicate a considerable difference (0.5 lbm/sec of oxidizer and 0.3 lbm/sec of fuel), between the propellant flow rates from like propellant tanks. Consideration of the engine gimbal angles and the feed line characteristics and configuration indicate that these differences in flow rates do not appear reasonable. The Propulsion Analysis Program calculated corresponding percentage rates of 0.174, 0.174, 0.175, and 0. 175 %/sec indicating that the Ox 2 and the Fu 2 gages best represented the actual flow rates.

With the exception of the Ox 2 gage, quantity readings during the portion of the burn analyzed appear to be within the expected accuracy of 1.3%. Table 5 presents a comparison of the Performance Analysis Program results to measured flight data. It is apparent from the Table that a bias of 1.2 percent would allow the Ox 2 gage to also be within the expected accuracy.

At ignition of the second and third DPS maneuvers, the propellant gages were in error, reading quantities of 10 to 50 percent more than were actually in the tanks. As the burns progressed, the readings approached the correct values but at engine shutdown the levels had not yet completely stabilized. This indicates that either the RCS ullage maneuver performed prior to DPS engine ignition was not sufficient to completely settle the propellants, or that at the relatively low vehicle acceleration conditions, propellant tended to cling to the gaging probes thus creating the erroneous output. The acceleration levels ranged from approximately two to six ft/sec^2 during these two burns. The maximum acceleration experienced during the Docked Burn was also approximately six ft/sec^2. The gaging system is designed to operate reliably at acceleration levels greater than 5.4 ft/sec^2. Thus, all DPS operation during Apollo 9 was near or less than the lower reliable operating limit of the gaging system.

Since the gaging system is not calibrated inside the propellant tanks at normal operating pressures, the analysis of the system is somewhat hampered. It also appears that neither repeatability nor carefully controlled dynamic tests of the system are performed. It is believed that a significant amount of uncertainty could be eliminated if such tests were carried out.

Propellant Loading

The propellant quantities loaded were as planned. Prior to actual loading, density determinations were made from three samples of each propellant to allow for correct off-loading of the planned overfill. An average oxidizer density of 90.25 lbm/ft^3 and an average fuel density of 56.46 lbm/ft^3 under a pressure of 240 psia and at a temperature of 70°F were determined from the sample. Propellant quantities of 11062.7 lbm of oxidizer at a temperature of 69.6°F and a pressure of 51.7 psig and 6977.2 lbm of fuel at a temperature of 70.5°F and a pressure of 29.2 psig were loaded. (Reference 10) The total quantity of propellant on board at launch was 18039.9 lbm.

ENGINE TRANSIENT ANALYSIS[1]

The mission duty cycle of the DPS during Apollo 9 included three starts at the minimum throttle setting, two shutdowns at approximately 40% throttle and one shutdown at the minimum throttle setting. Manual throttling to several different thrust settings was performed during the first DPS maneuver and manual throttling from minimum thrust to approximately 40% thrust was accomplished during the second DPS maneuver.

Combustion Disturbances

Due to the blockage in the internal heat exchanger of the SHe tank, the DPS essentially operated in a blowdown mode for the initial portion of the docked burn. Because of the low interface pressures arising from this situation, the engine experienced rougher than normal combustion at FTP. The chamber pressure experienced oscillations of approximately ± 8 psia compared to approximately ± 4 psia later in the FTP portion of the burn when system pressures were nominal. Also, during the early portion of FTP operation, four combustion disturbances (or "pops") of greater than 20 psia were observed. Two pops occurred near the end of the FTP portion. All disturbances damped out in less than 0.01 seconds. The rough combustion has been observed during ground tests and is considered characteristic of the engine under the system conditions experienced. The pops have also been observed during ground tests and are considered characteristic of the engine.

Start and Shutdown Transients

The ignition delay from fire switch (FS-1) to first rise in chamber pressure was approximately 1.4, 0.7 and 0.7 for the first, second and third starts respectively. Engine ignition delay is a function of priming

[1]Reference 6

conditions, engine valve response time and pressure levels. The pressure levels and valve response time were approximately equal for all three burns. The difference in time between the first and subsequent starts appears to be because of a difference in priming conditions. The first burn was with no propellant between the engine prevalve (actuator isolation valves) and the engine valve actuators, and no propellant between the series engine shutoff valves. These areas were primed with liquid for the second and third burn. Similar observations were made relating to starts performed during the Apollo 5/LM-1 Mission.

A summary of the start and shutdown transients for the first, second and third DPS maneuvers is presented in Table 6.

The transient time from FS-1 to 90% of steady state thrust for minimum thrust starts were within the specification limit of 4.0 seconds. There are no specifications that relate to the time for shutdowns performed from throttle positions other than FTP. Reference 10 indicates that the expected nominal decay time is 0.6 and 4.0 seconds for 40% and 12% shutdowns respectively. In the three sigma maximum case the decay times are expected to be 1.1 and 7.0 seconds. It can be seen in Table 6 that shutdown transient times were within these expected limits.

There is no specification limit for total impulse during start or shutdowns. There is, however, a repeatability requirement of ± 100 lbf-sec. The repeatability requirement was not met. However, this requirement should only apply to starts and shutdowns to and from FTP. Ground tests have shown that the impulse during low thrust starts and shutdowns are less repeatable than during FTP transients. Based on ground test results, it is felt that the start and shutdown impulse repeatability was satisfactory.

Throttle Response

During the Docked Burn, the engine was commanded to several different throttle settings to demonstrate adequate engine throttle response. Except for throttling to and from FTP, all throttling was executed by manual commands. The manual throttle changes were made relatively slowly, being performed as throttling ramps over approximately two to three seconds while the throttling to and from FTP were very rapid, caused by step changes in engine voltage commanded by the LM Guidance Computer. Table 7 presents the throttle changes executed during this burn. The engine throttle response appeared nominal with the engine chamber pressure responding within 0.15 seconds after the initiation of a throttling command. During the slow manual throttling, the chamber pressure tracked the commands very well, reaching steady state operation at the next throttle position with no discernible lag compared to engine voltage. Table 8 presents a summary of the total response time for each of the commands executed during the burn.

The throttling performed during the docked burn cannot be clearly compared with specification requirements. There are two specifications: 1) minimum to maximum thrust within 1.0 seconds, and 2) maximum to 50% thrust within 0.5 seconds. These requirements are for step command signal to the engine. No throttling performed during Apollo 9 were over the specified throttling range although the throttling to and from FTP was similar. In both of these cases, the specification response time limits were not exceeded.

The throttle response time for the second DPS burn was not analyzed since 1) the throttle change was performed very slowly and 2) the helium ingestion which occurred during this burn made meaningful analysis difficult.

REFERENCES

1. TRW IOC 69.4354.2-28, "DPS Input to Apollo 9 Mission Report", from R.K.M. Seto to D. W. Vernon, 9 April 1969.

2. SPD8-R-005, "Mission Requirement D Type Mission, LM Evaluation and Combined Operations", 21 January 1969.

3. NASA/MSC Report MSC-PA-R-69-2, "Apollo 9 Mission Report, May 1969.

4. TRW IOC 68.4716.2-12, "Apollo Mission D/LM-3/DPS Preflight Performance Report - Revision", from R.K.M. Seto, D. F. Rosow and S. C. Wood to D. W. Vernon, 26 November 1968.

5. TRW Report, "Apollo 9 CSM 104 Service Propulsion System Final Flight Evaluation", R. J. Smith, June 1969.

6. MSC Memorandum EP22-41-69, "Transient Analysis of Apollo 9 LMDE", from EP2/Systems Analysis Section to EP2/Chief, Primary Propulsion Branch, 5 May 1969.

7. TRW Report 01827-6241-TO-00, "Review of the LM-3, DPS-2, Roughness Anomaly", J. A. Katherler, 25 April 1969.

8. Airesearch Report 69-4976, "Analysis of Performance Anomalies During LM-3 Mission of Supercritical Helium Storage and Supply System Descent Propulsion for Lunar Module (LM)", A. W. Keeley, 21 April 1969.

9. GAEC LM-3 Quick-Look Anomaly Report.

10. SNA-8-D-027 (III) Rev. 1, "CSM/LM Spacecraft Operational Data Book", Vol. III, Mass Properties, November 1968.

TABLE 1

DPS MISSION DUTY CYCLE

BURN	FS-1 (hr:min:sec)	FS-2 (hr:min:sec)	BURN DURATION (secs)	VELOCITY CHANGE (ft/sec)
DPS 1	49:41:34.7[(2)]	49:47:44.4[(2)]	369.7	1741.7
DPS 2	93:47:34.4[(1)]	93:47:54.3	19.7	89.3
DPS 3	95:39:08.3[(2)]	95:39:30.6[(2)]	22.3	42.2

[(1)] Times are from LM Guidance Computer Downlink Data - GG0001X

[(2)] Reference 6

TABLE 2

LM-3 DPS ENGINE AND FEED SYSTEM PHYSICAL

CHARACTERISTICS

ENGINE	
Engine Number	1030
Chamber Throat Area, In^2	53.962[1]
Nozzle Exit Area, In^2	2569.7[4]
Nozzle Expansion Ratio	47.6[4]
Oxidizer Interface To Chamber Resistance At FTP $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	3934.0[3]
Fuel Interface to Chamber Resistance at FTP $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	6220.0
FEED SYSTEM	
Oxidizer Propellant Tanks, Total Ambient Volume, ft^3	126.0[4]
Fuel Propellant Tanks, Total Ambient Volume, ft^3	126.0[4]
Oxidizer Tank to Interface Resistance, $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	455.11[2]
Fuel Tank to Interface Resistance, $\frac{lbm\text{-}sec^2}{lbf\text{-}ft^5}$	661.70[2]

[1] TRW No. 01827-6122-T000, TRW LM Descent Engine Serial No. 1030, Acceptance Test Performance Report Paragraph 6.9, 11 December 1967.

[2] GAEC Cold Flow Tests

[3] TWX-MSG04900, 20 March 1968, from R. D. Baker, TRW/LA "Hydraulic Resistances for LMDE S/N 1030."

[4] Approximate Values

TABLE 3

FLIGHT DATA USED IN STEADY STATE ANALYSIS

MEASUREMENT NUMBER	DESCRIPTION	RANGE	SAMPLE RATE SAMPLE/SEC
GQ3611P	Pressure, Engine Fuel Interface	0-300 psia	Continuous
GQ4111P	Pressure, Engine Oxidizer Interface	0-300 psia	1
GQ3666P	Pressure Difference, Fuel Tank Bottom to Interface	0-35 psid	Continuous
GQ4116P	Pressure Difference, Oxidizer Tank Bottom to Interface	0-35 psid	Continuous
GQ6501	Pressure, Fuel Injector Inlet	0-300 psia	Continuous
GQ6506P	Pressure, Oxidizer Injector Inlet	0-300 psia	Continuous
GQ3603Q	Quantity, Fuel Tank No. 1	0-95 percent	1
GQ3604Q	Quantity, Fuel Tank No. 2	0-95 percent	1
GQ4103Q	Quantity, Oxidizer Tank No. 1	0-95 percent	1
GQ4104Q	Quantity, Oxidizer Tank No. 2	0-95 percent	1
GQ3718T	Temperature, Fuel Bulk Tank No. 1	20-120° F	1
GQ3719T	Temperature, Fuel Bulk Tank No. 2	20-120° F	1
GQ3811T	Temperature, Fuel Interface	0-200° F	Continuous
GQ4218T	Temperature, Oxidizer Bulk Tank No. 1	20-120° F	1
GQ4219T	Temperature, Oxidizer Bulk Tank No. 2	20-120° F	1
GQ4311T	Temperature, Oxidizer Interface	0-200° F	Continuous
GG0001X	PGNS Downlink Data	40 Bits	1/2

TABLE 4

DESCENT PROPULSION SYSTEM STEADY STATE PERFORMANCE

DOCKED DPS BURN

PARAMETER	FS-1 + 55 SECONDS			FS-1 + 215 SECONDS		
	PREDICTED	MEASURED	CALCULATED	PREDICTED	MEASURED	CALCULATED
INSTRUMENTED						
Regulator Outlet Pressure, psia	247	243	--	247	243	--
Oxidizer Delta Pressure, psid (Tank Bottom to Interface)	13.6	13.7	13.7	14.0	14.0	14.0
Fuel Delta Pressure, psid (Tank Bottom to Interface)	12.7	12.6	12.7	13.0	12.9	12.9
Oxidizer Interface Pressure, psia	225	222	222	225	221	221
Fuel Interface Pressure, psia	226	222	222	226	222	222
Oxidizer Injector Inlet Pressure, psia	215	212	211	214	210	210
Fuel Injector Inlet Pressure, psia	131	129	128	129	124	125
Engine Chamber Pressure, psia	106	107	103	103	108	100
Oxidizer Bulk Temperature, °F	66	69	--	66	69	--
Fuel Bulk Temperature, °F	66	69	--	66	69	--
Oxidizer Interface Temperature, °F	66	69	--	66	69	--
Fuel Interface Temperature, °F	61	65	--	61	65	--
DERIVED						
Oxidizer Flowrate, lbm/sec	19.9	--	19.8	20.1	--	20.0
Fuel Flowrate, lbm/sec	12.6	--	12.5	12.7	--	12.6
Propellant Mixture Ratio	1.58	--	1.58	1.58	--	1.58
Vacuum Specific Impulse, sec	303.3	--	303.0	302.6	--	301.5
Vacuum Thrust, lbf	9847	--	9784	9919	--	9831
Throat Erosion, %	.14	--	.14	3.09	--	4.18

TABLE 5

DPS PROPELLANT QUANTITY GAGING SYSTEM PERFORMANCE

Parameter	Time, hr:min:sec			
	49:42:40	49:43:30	49:44:20	49:45:10
Oxidizer Tank No. 1				
Measured Quantity, percent	88.9	79.8	71.0	62.0
Calculated Quantity, percent	88.1	79.4	70.7	62.0
Difference, percent	+0.8	+0.4	+0.3	+0.0
Oxidizer Tank No. 2				
Measured Quantity, percent	90.0	81.6	73.2	64.1
Calculated Quantity, percent	88.1	79.4	70.7	62.0
Difference, percent	+1.9	+2.2	+2.5	+2.1
Fuel Tank No. 1				
Measured Quantity, percent	89.5	80.5	71.7	62.5
Calculated Quantity, percent	88.8	80.0	71.3	62.5
Difference, percent	+0.7	+0.5	+0.4	+0.0
Fuel Tank No. 2				
Measured Quantity, percent	88.5	79.9	71.5	62.5
Calculated Quantity, percent	88.8	80.0	71.3	62.5
Difference, percent	-0.3	-0.1	+0.2	+0.0

TABLE 6

DPS START AND SHUTDOWN IMPULSE

SUMMARY

	Docked Burn DPS-1	Phasing Burn DPS-2	Insertion Burn DPS-3
Total vacuum start impulse (FS-1 to 90% steady state), lbf-sec	805	1029	950
Steady State Thrust Level	Minimum	Minimum	Minimum
Start Time (FS-1 to 90% steady state), sec	2.5[1]	2.1	2.3[1]
Total vacuum shutdown impulse (FS-2 to 10% steady state), lbf-sec	---[3]	1730	748[2]
Steady State Thrust Level	40%	40%	Minimum
Shutdown Time (FS-2 to 10% steady state), sec	1.1[1]	1.1	1.8[1]
Total vacuum shutdown impulse (FS-2 to zero thrust) from velocity gain data , lbf-sec	1777	---	1040

[1] Reference 6

[2] Pc decay data to 17.5% of steady state only

[3] Pc decay data was unavailable

TABLE 7

THROTTLING COMMANDS

DOCKED BURN

COMMANDED THROTTLE CHANGES		TIME OF INITIATION OF THROTTLE CHANGES	
PLANNED	ACTUAL	HR:MIN:SEC GET	SECONDS GET
Minimum to 40%	12.7% to 37.9%	49:41:41.70 ± 0.05	178901.70 ± 0.05
40% to FTP	37.9% to FTP	49:42:00.68 ± 0.02	178920.68 ± 0.02
FTP to 40%	FTP to 37.9%	49:46:48.53 ± 0.02	179208.53 ± 0.02
40% to Minimum	37.9% to 12.7%	49:46:53.15 ± 0.05	179213.15 ± 0.02
Minimum to 40%	12.7% to 38.3%	49:47:01.58 ± 0.02	179221.58 ± 0.02
40% to 25%	38.3% to 22.5%	49:47:08.56 ± 0.02	179228.56 ± 0.02
25% to 40%	22.5% to 36.8%	49:47:14.95 ± 0.02	179234.95 ± 0.02

TABLE 8

THROTTLE RESPONSE TIME

DOCKED BURN

Commanded Throttle Changes	Time From Command To Engine Steady State Operation, Sec.	Planned Time For Throttle Change, Sec.	Mode Of Throttle Change
12.7% to 37.9%	2.40 ± 0.10	Immediate (Step Change)	Manual
37.9% to FTP	0.82 ± 0.05	Immediate (Step Change)	Auto
FTP to 37.9%	0.50 ± 0.05	Immediate (Step Change)	Auto
37.9% to 12.7%	3.00 ± 0.10	5.0	Manual
12.7% to 38.3%	3.35 ± 0.0	5.0	Manual
38.3% to 22.5%	4.10 ± 0.10	2.0	Manual
22.5% to 36.8%	3.05 ± 0.10	2.0	Manual

Figure 1

DPS1 - Docked Burn Throttling Profile

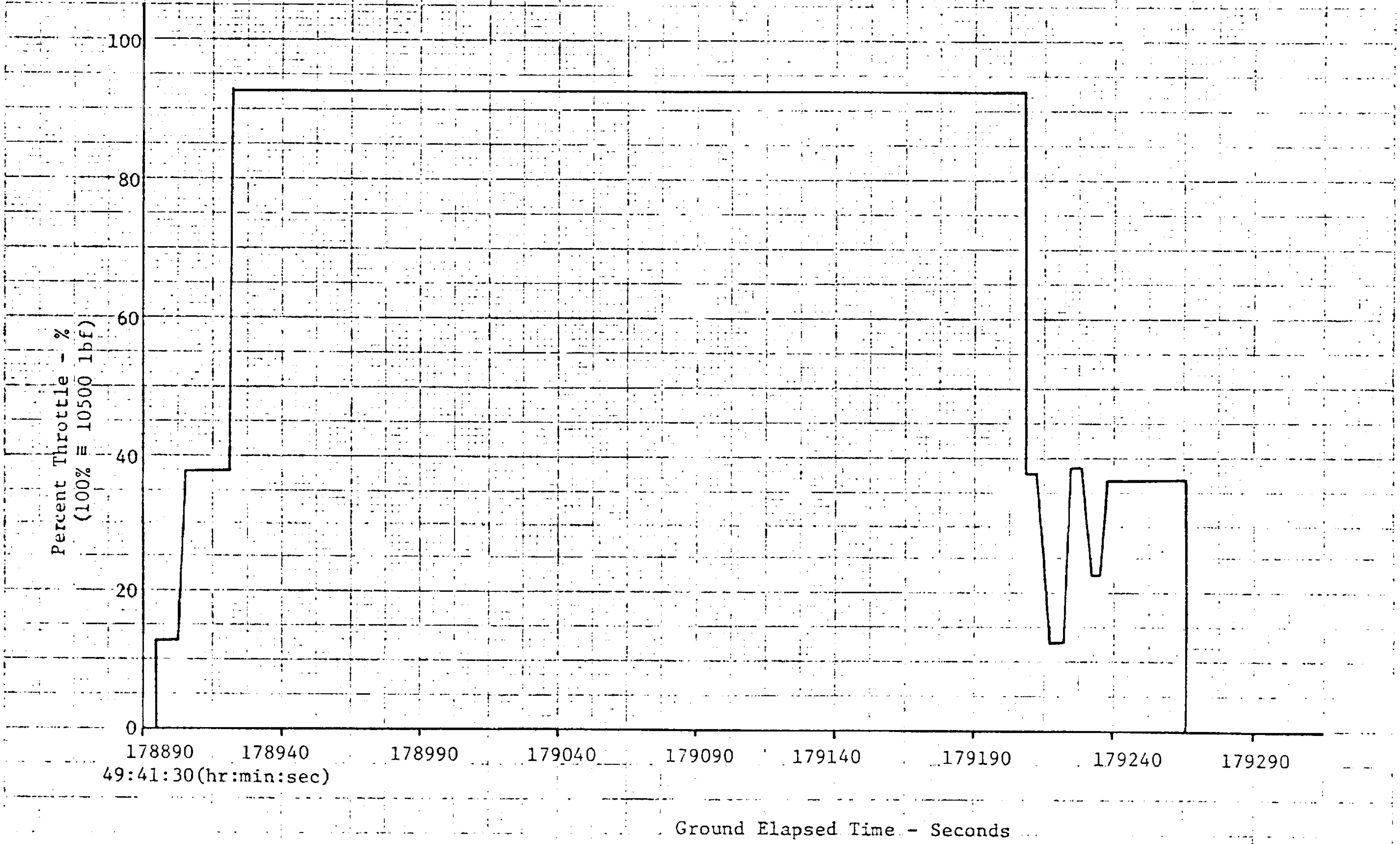
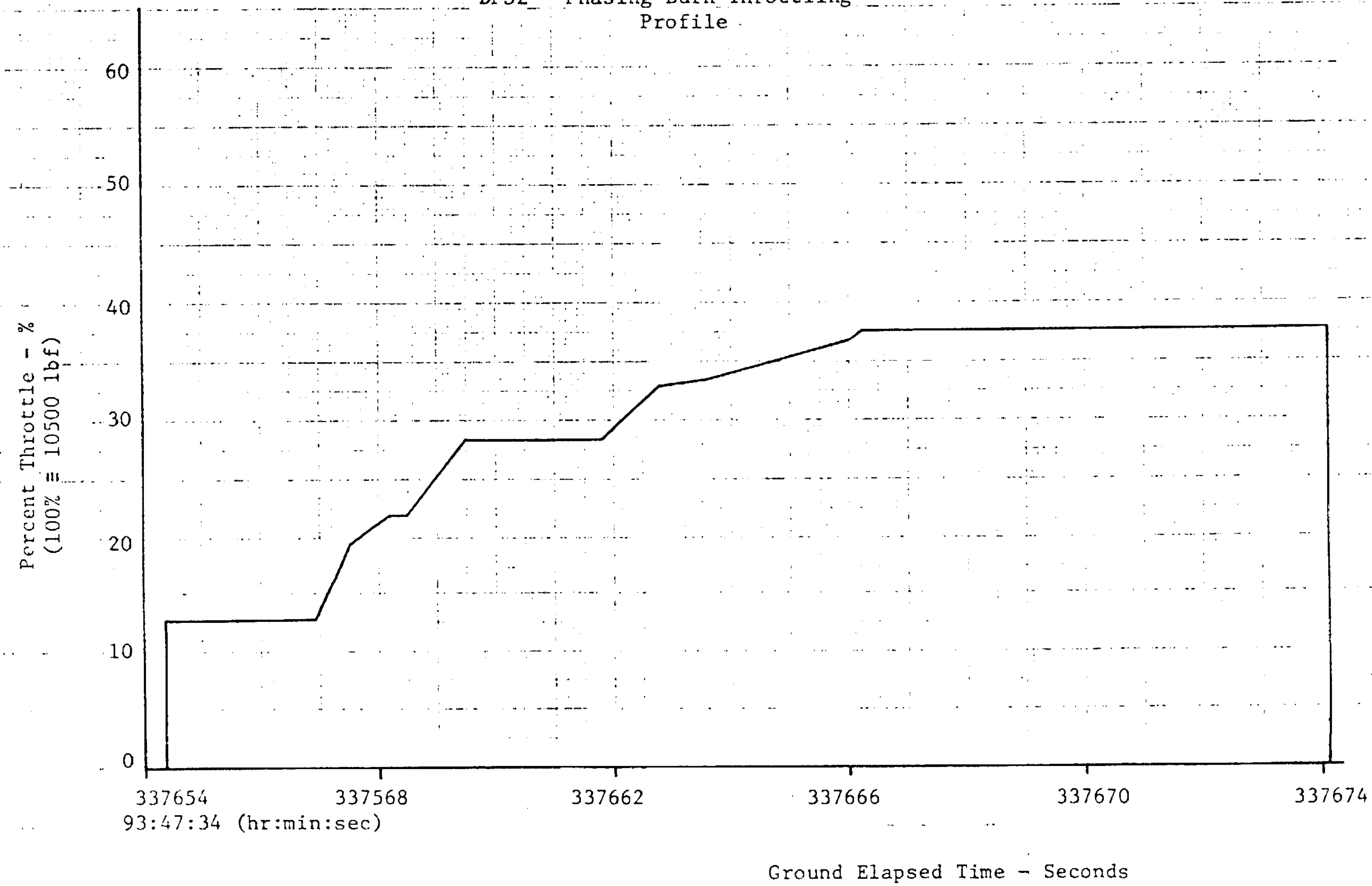


Figure 2

DPS2 - Phasing Burn Throttling Profile



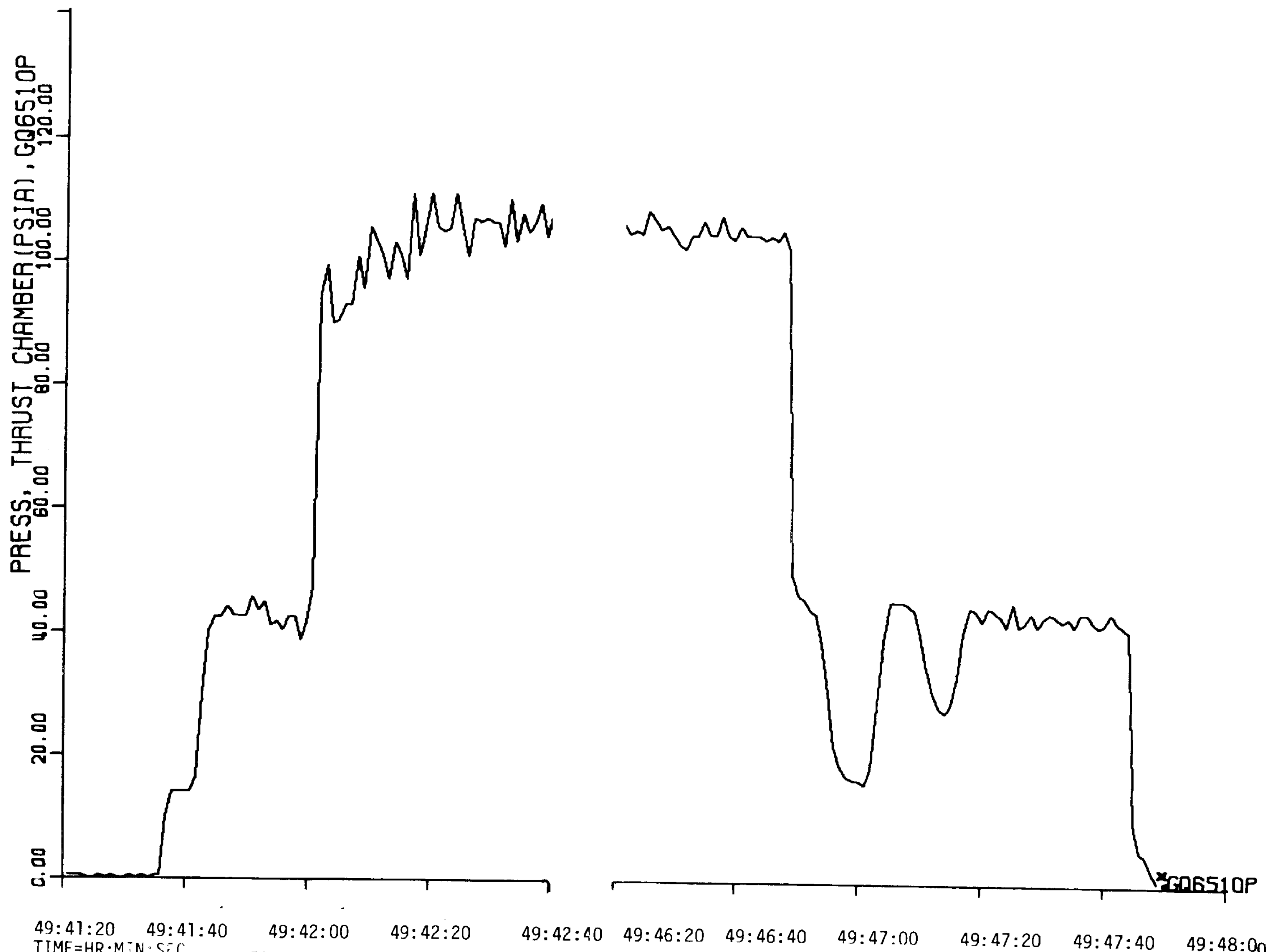


FIG. 3 — Chamber Pressure During First Descent Propulsion Firing

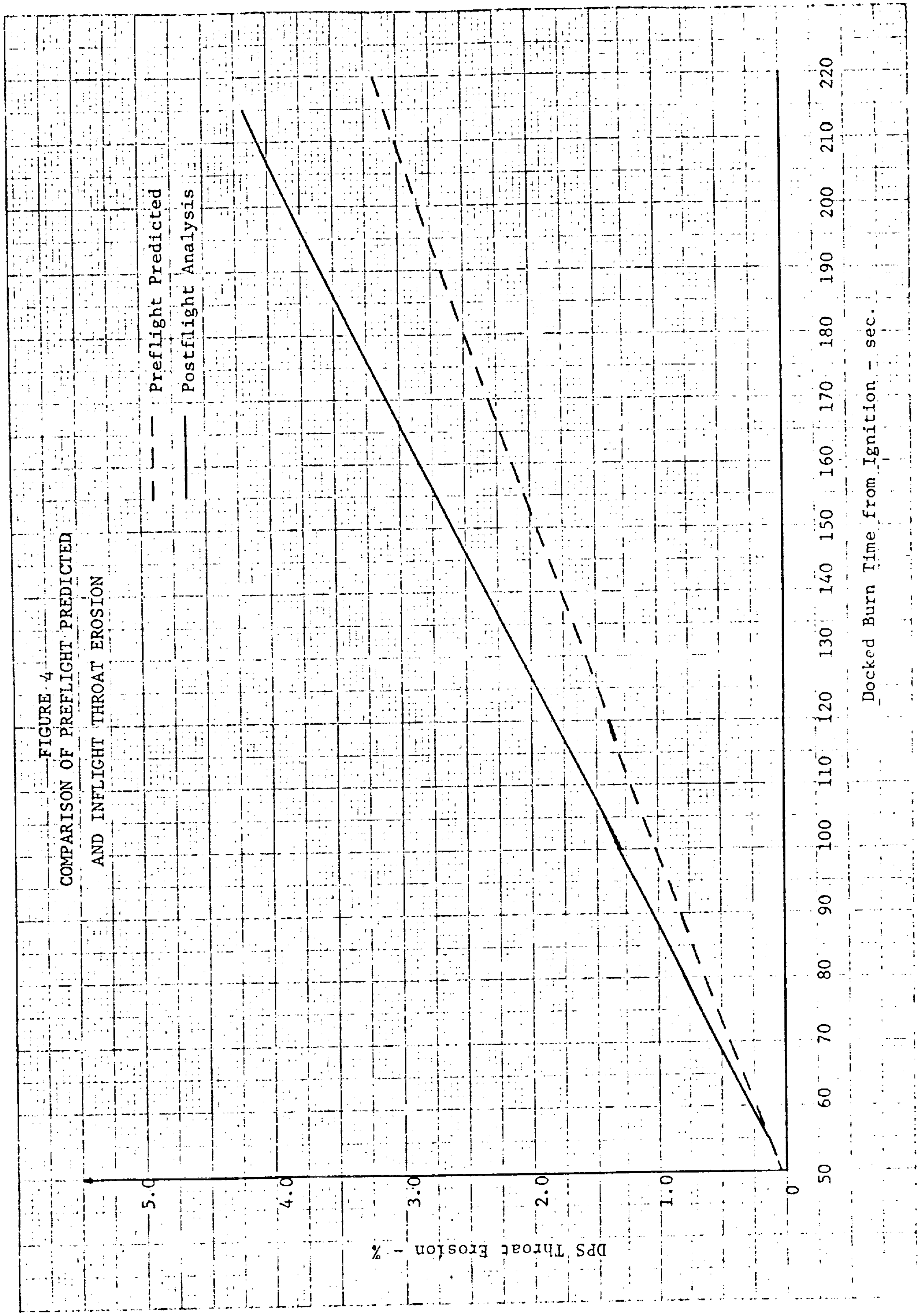


FIGURE 4
COMPARISON OF PREFLIGHT PREDICTED
AND INFLIGHT THROAT EROSION

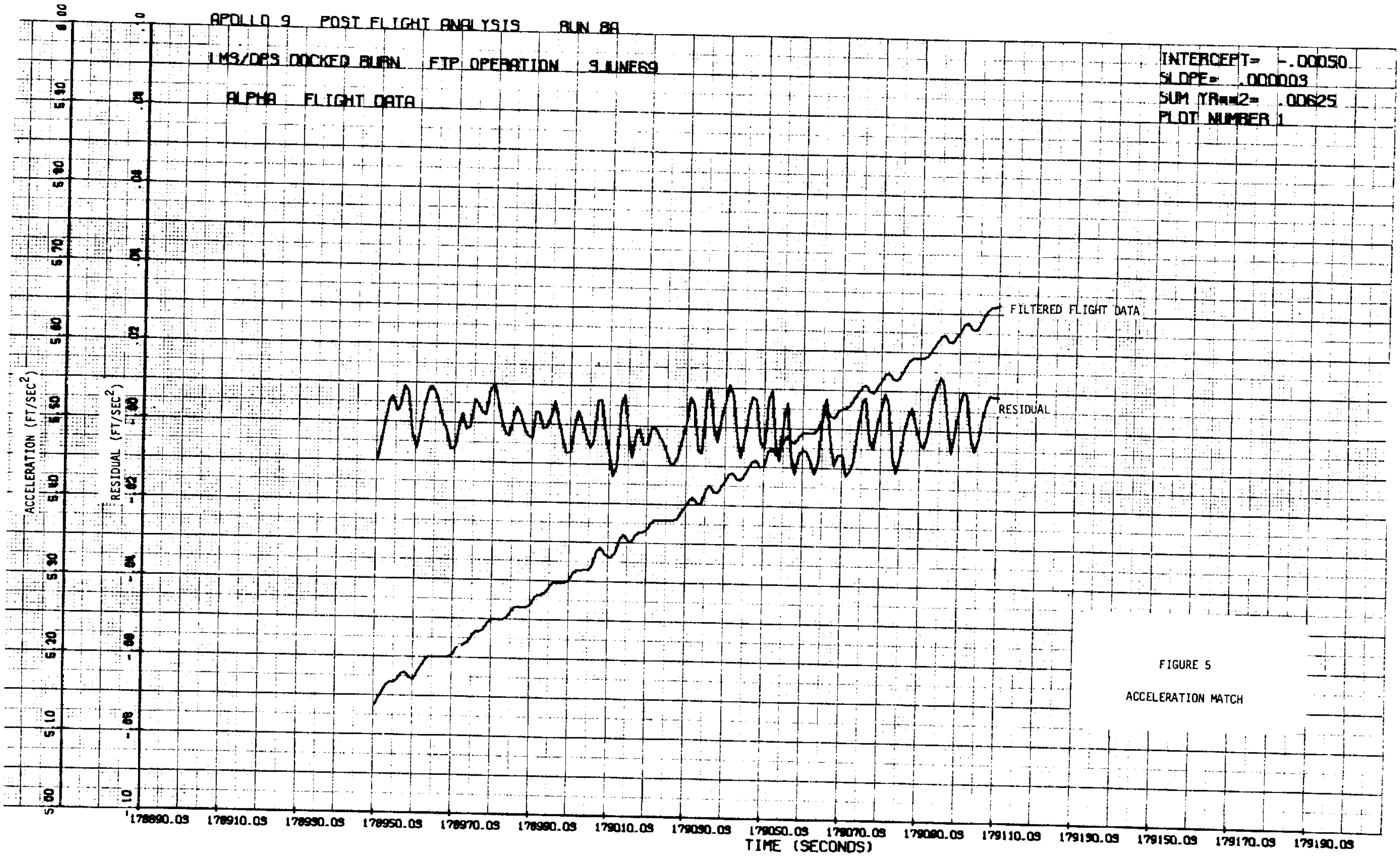


FIGURE 5

ACCELERATION MATCH

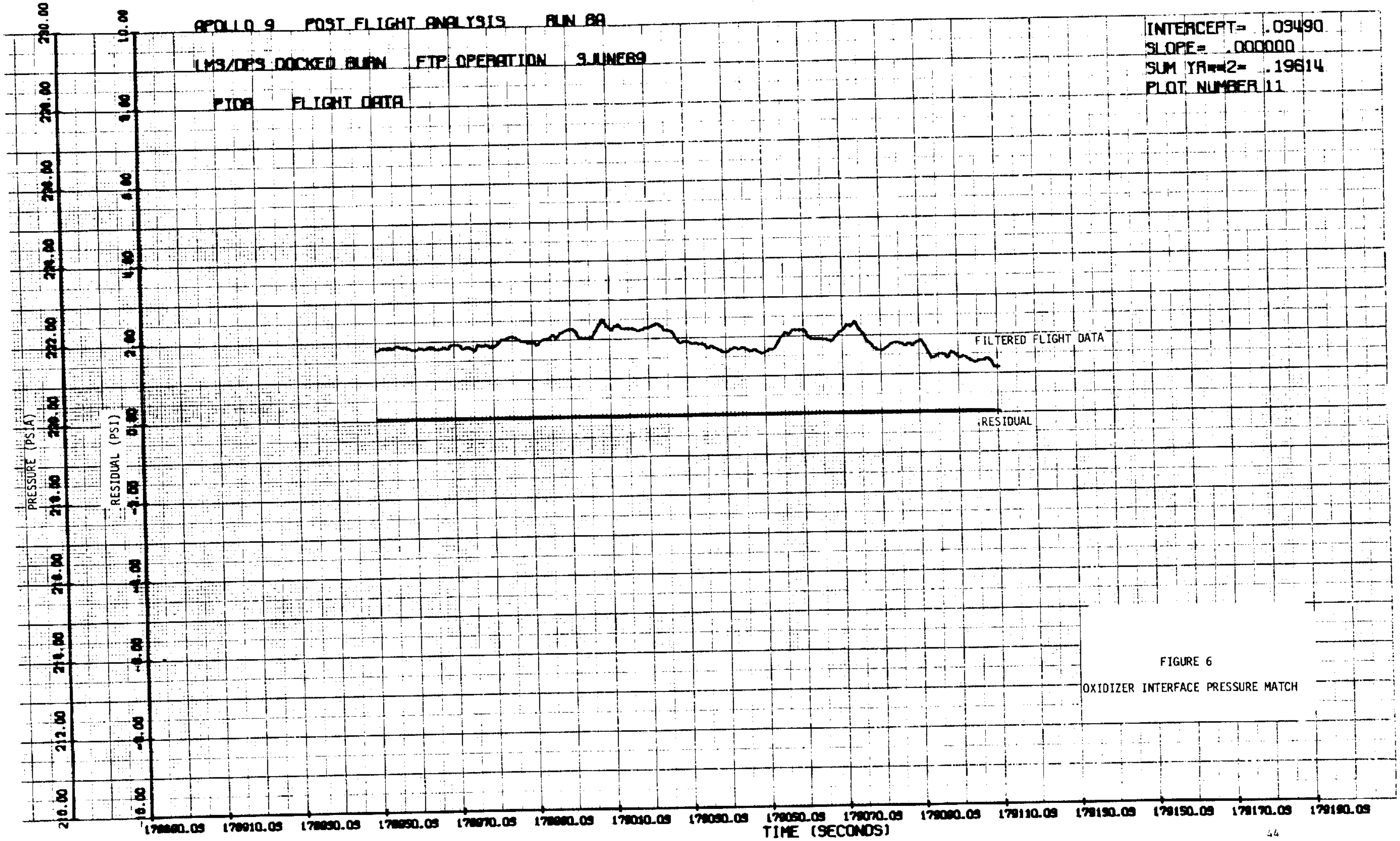


FIGURE 6

OXIDIZER INTERFACE PRESSURE MATCH

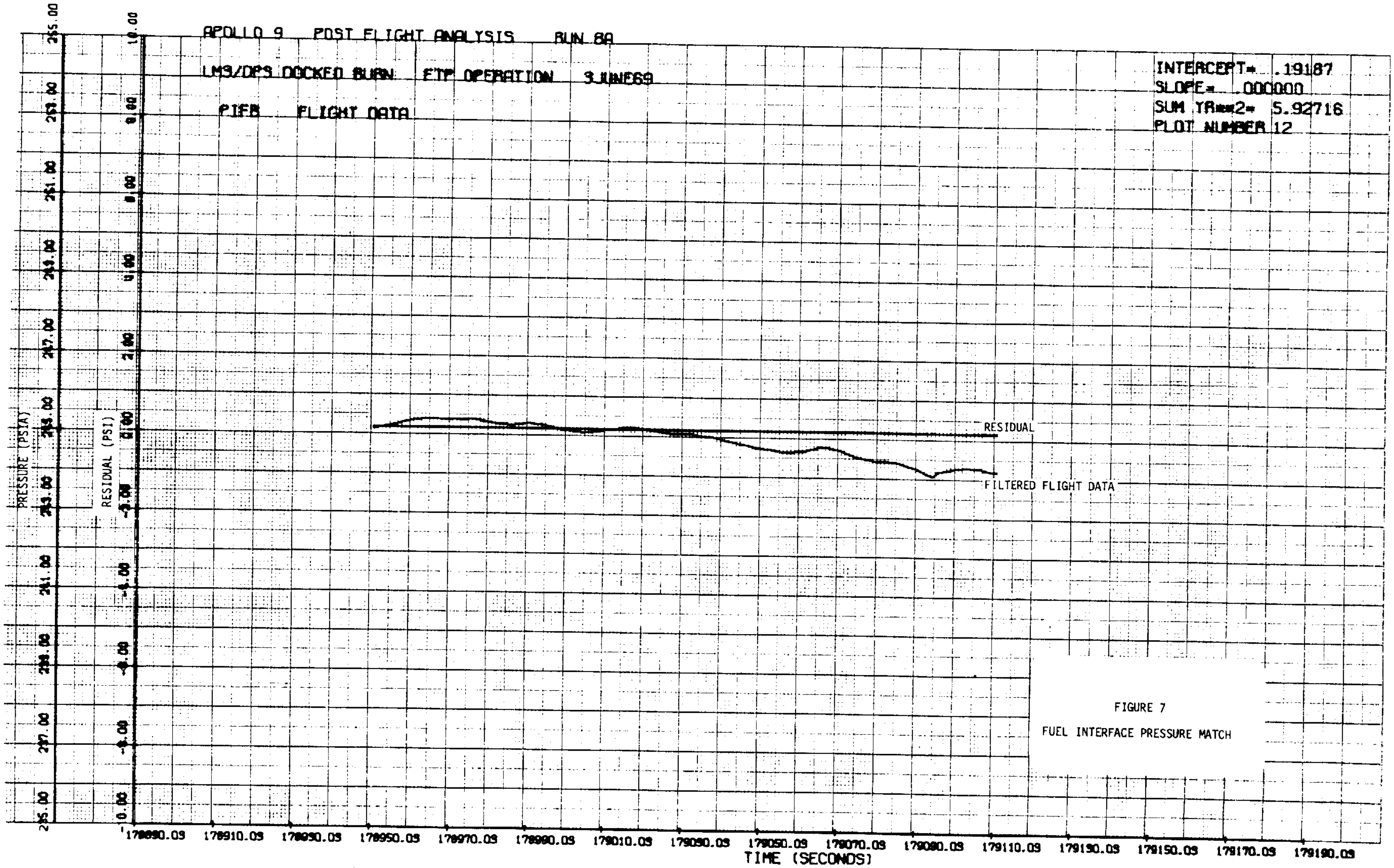


FIGURE 7
FUEL INTERFACE PRESSURE MATCH

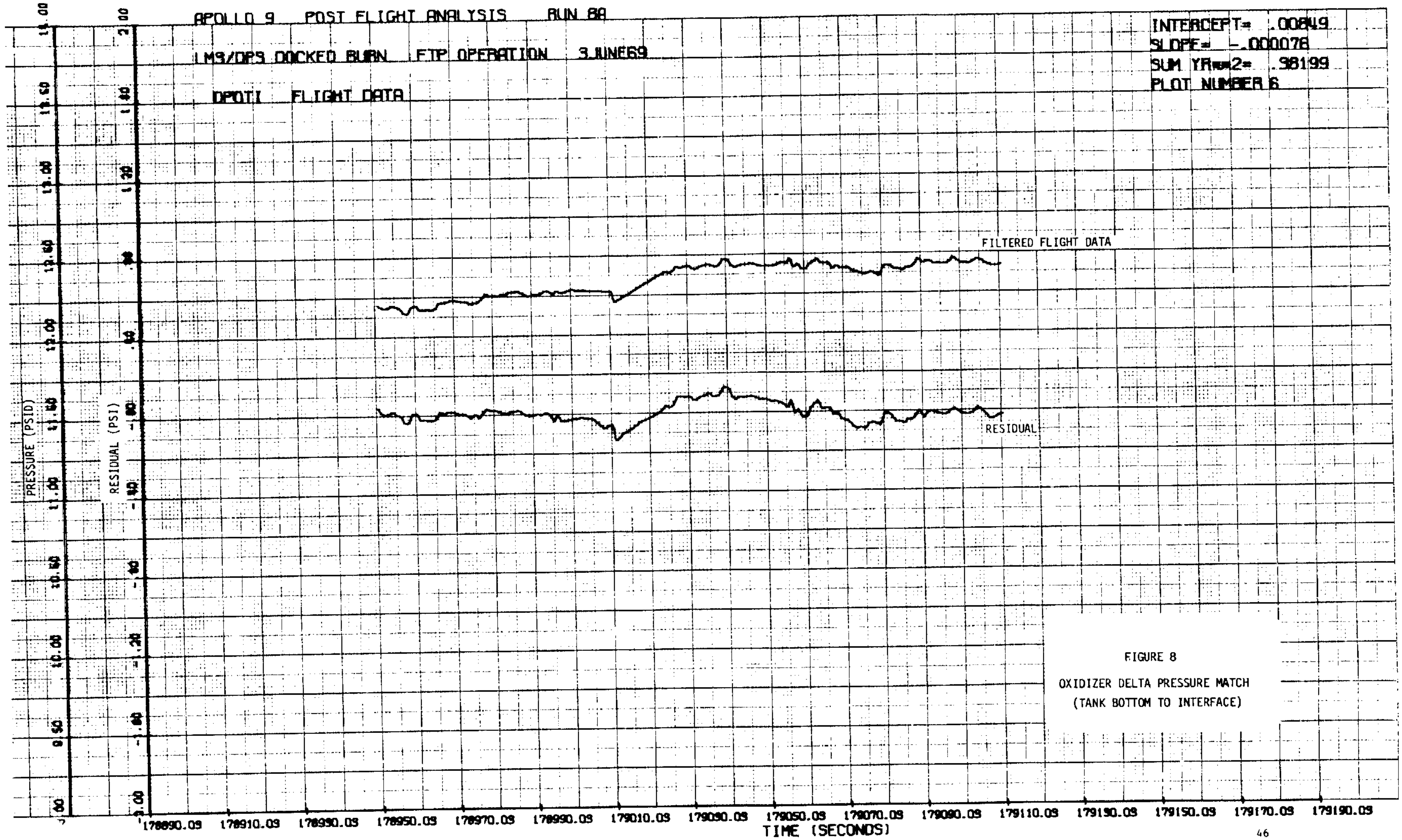


FIGURE 8

OXIDIZER DELTA PRESSURE MATCH
(TANK BOTTOM TO INTERFACE)

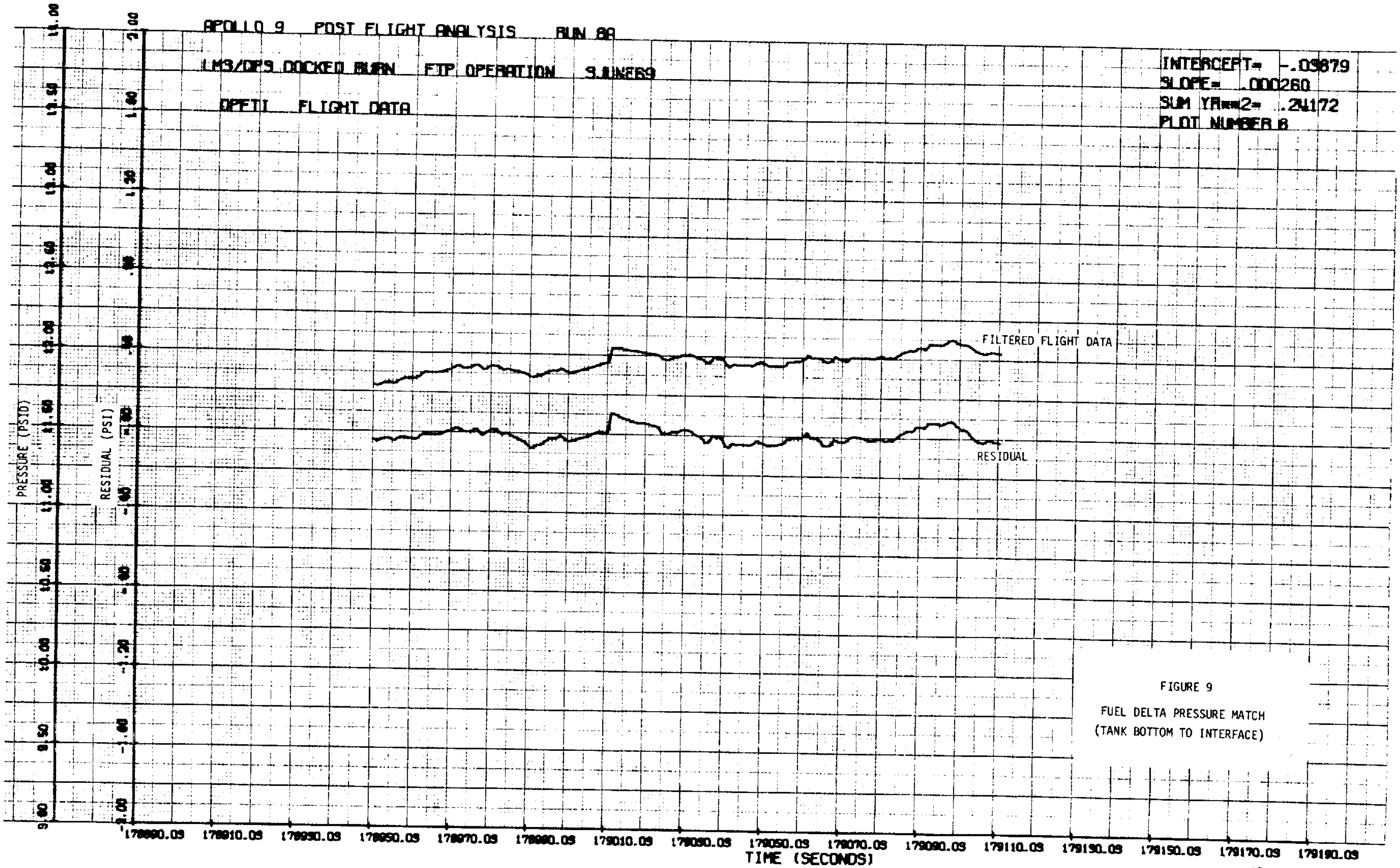


FIGURE 9

FUEL DELTA PRESSURE MATCH
(TANK BOTTOM TO INTERFACE)

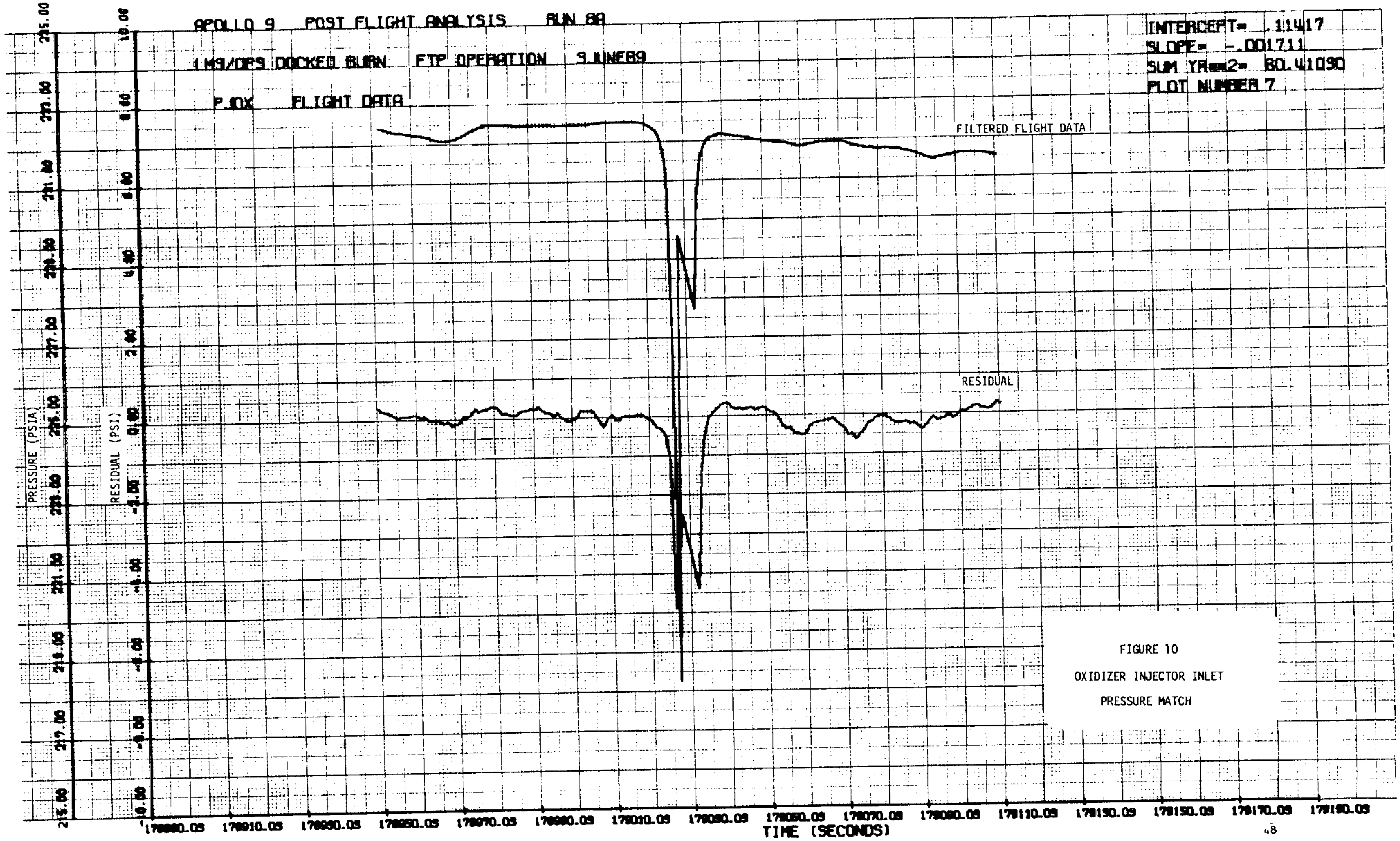


FIGURE 10

OXIDIZER INJECTOR INLET

PRESSURE MATCH

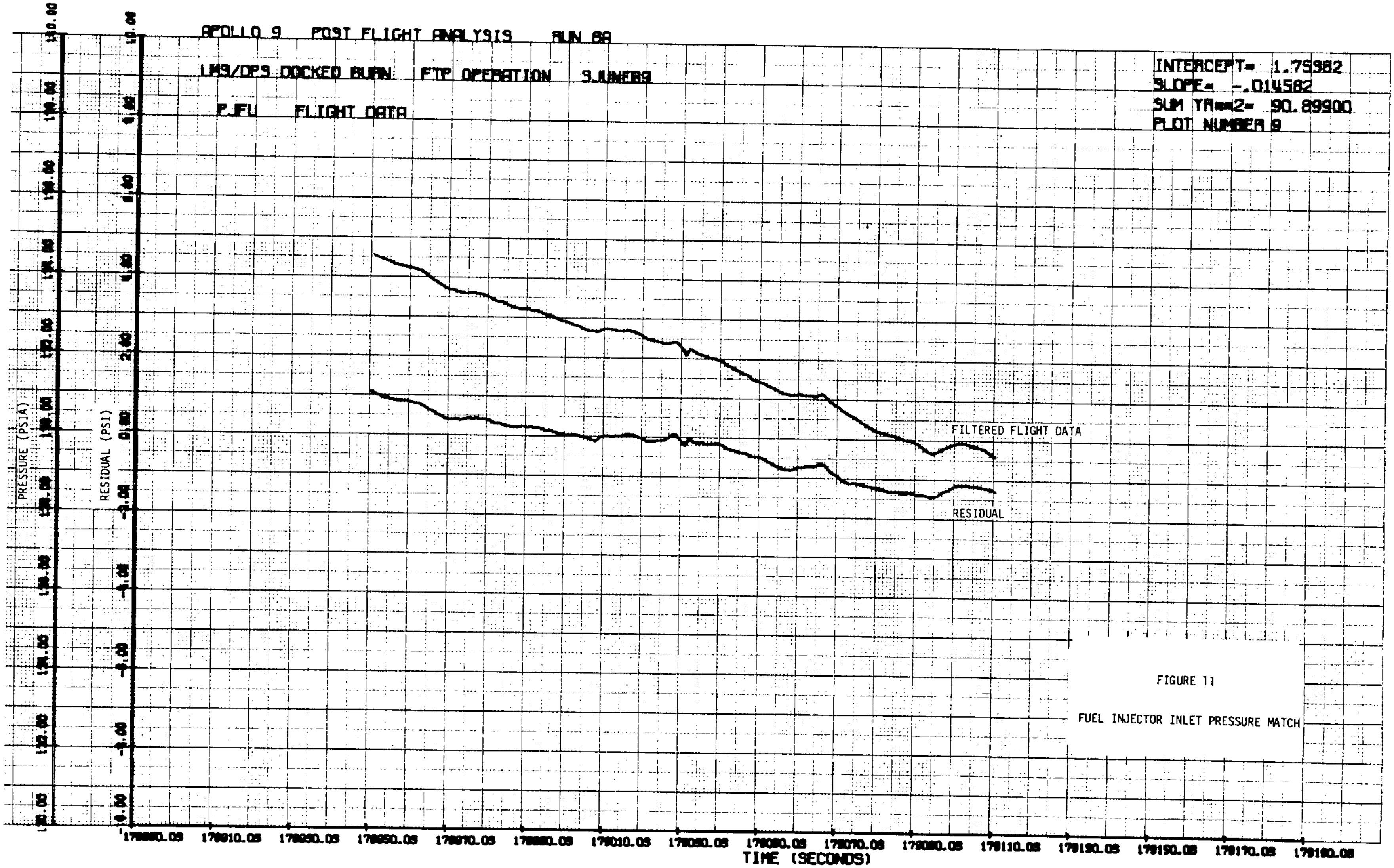


FIGURE 11

FUEL INJECTOR INLET PRESSURE MATCH

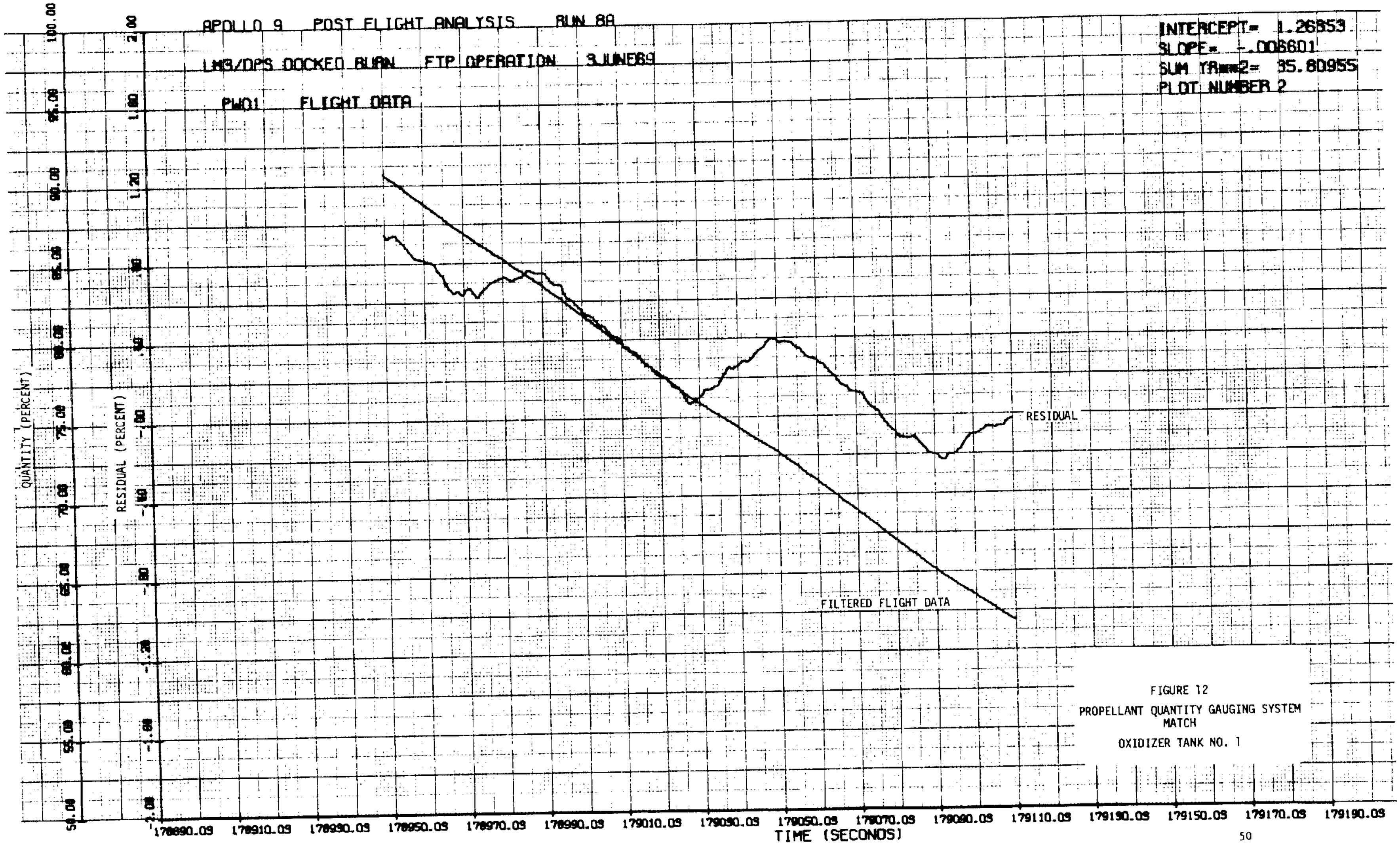


FIGURE 12
PROPELLANT QUANTITY GAUGING SYSTEM MATCH
OXIDIZER TANK NO. 1

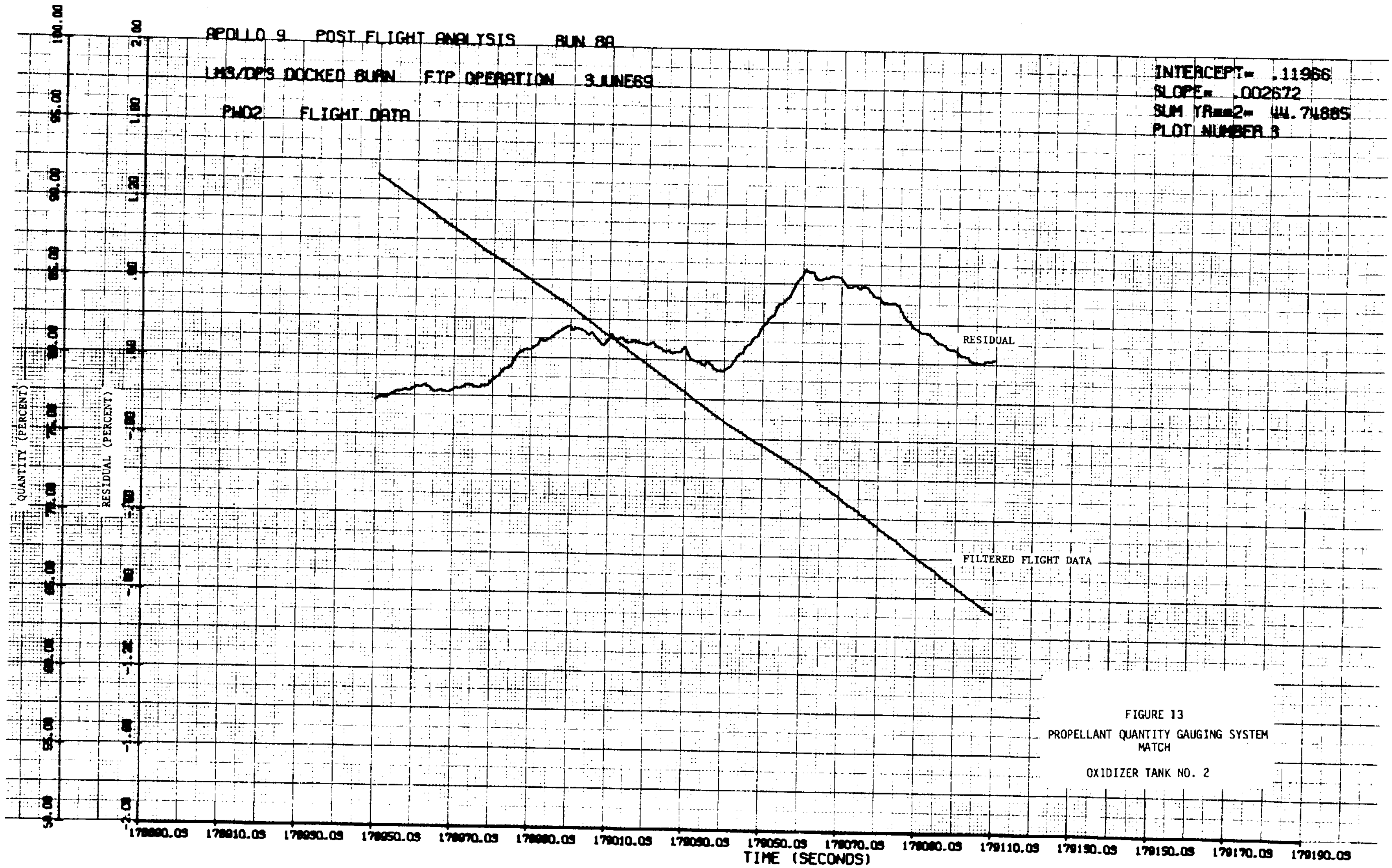


FIGURE 13
PROPELLANT QUANTITY GAUGING SYSTEM MATCH
OXIDIZER TANK NO. 2

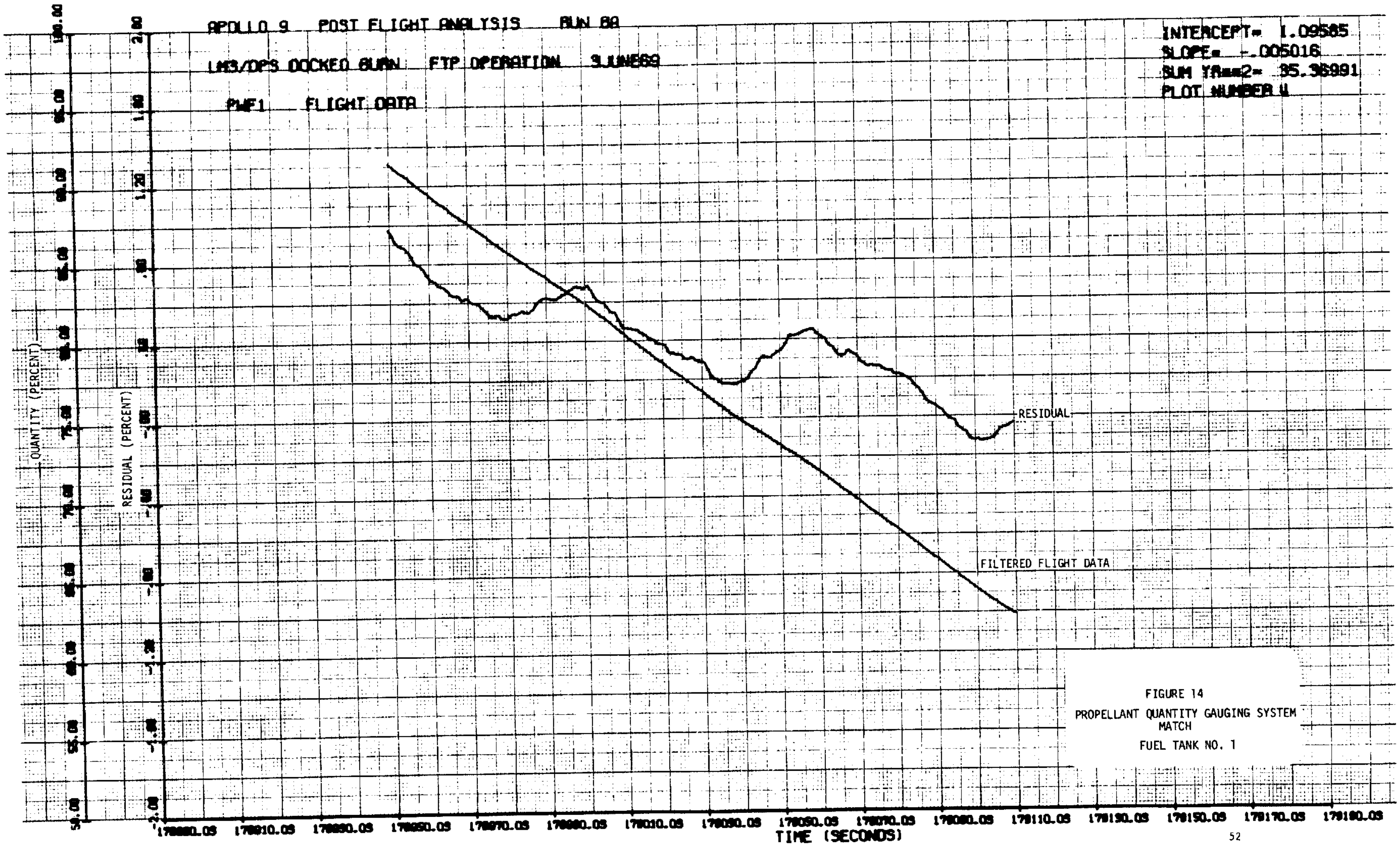


FIGURE 14
PROPELLANT QUANTITY GAUGING SYSTEM MATCH
FUEL TANK NO. 1

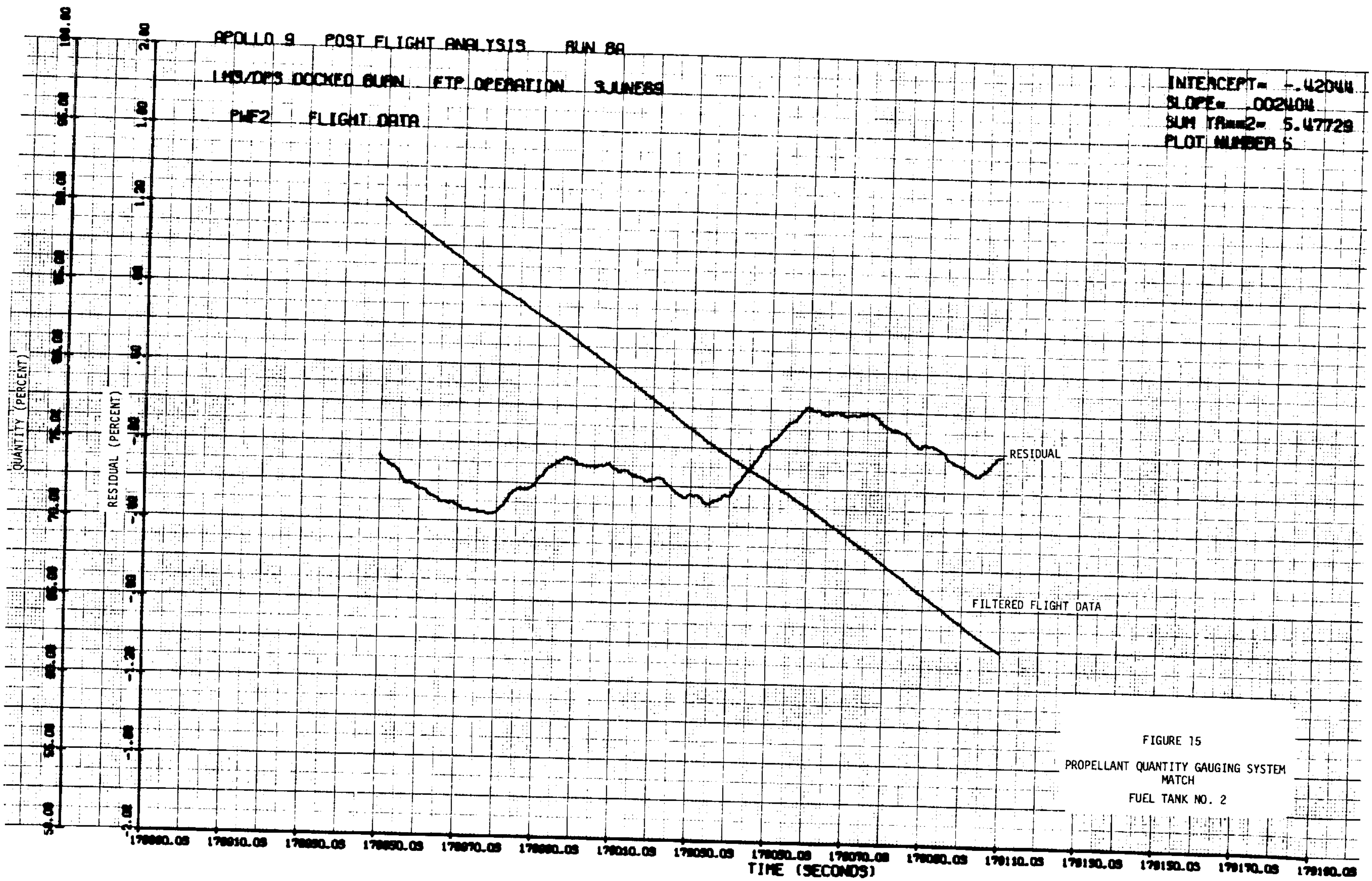


FIGURE 15

PROPELLANT QUANTITY GAUGING SYSTEM MATCH

FUEL TANK NO. 2

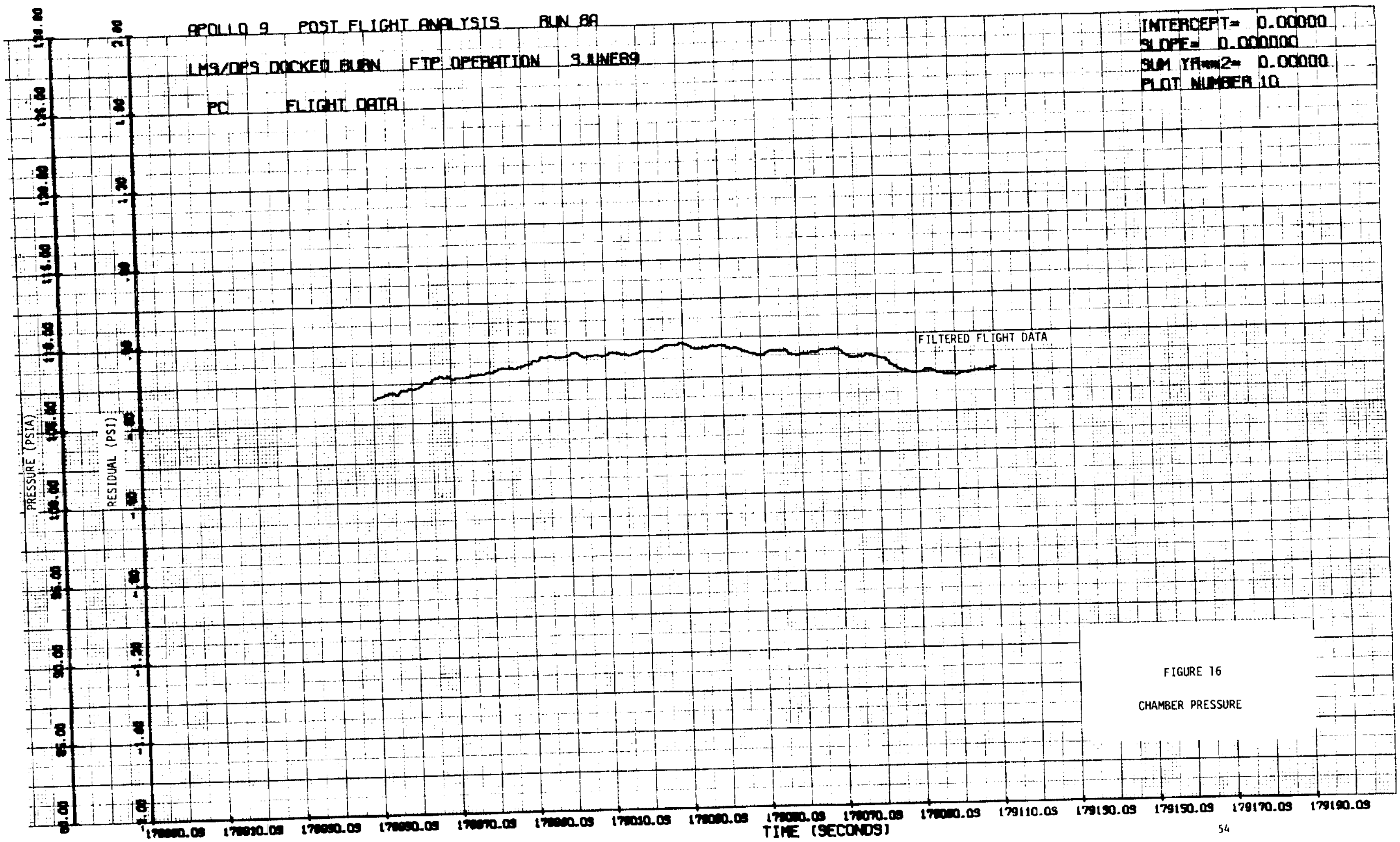


FIGURE 16

CHAMBER PRESSURE

FIGURE 17

COMPARISON OF PREFLIGHT PREDICTED AND INFLIGHT PERFORMANCE

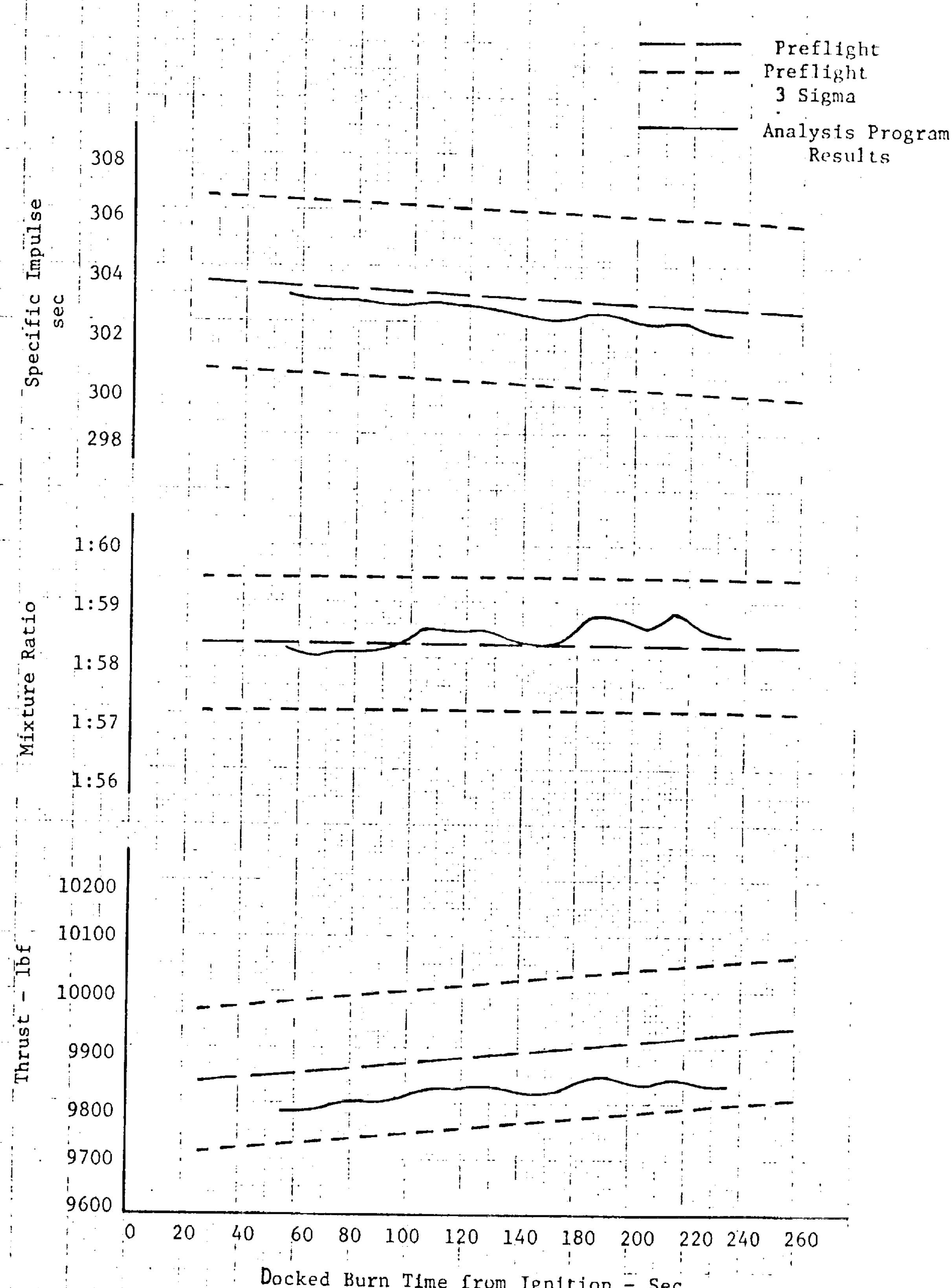
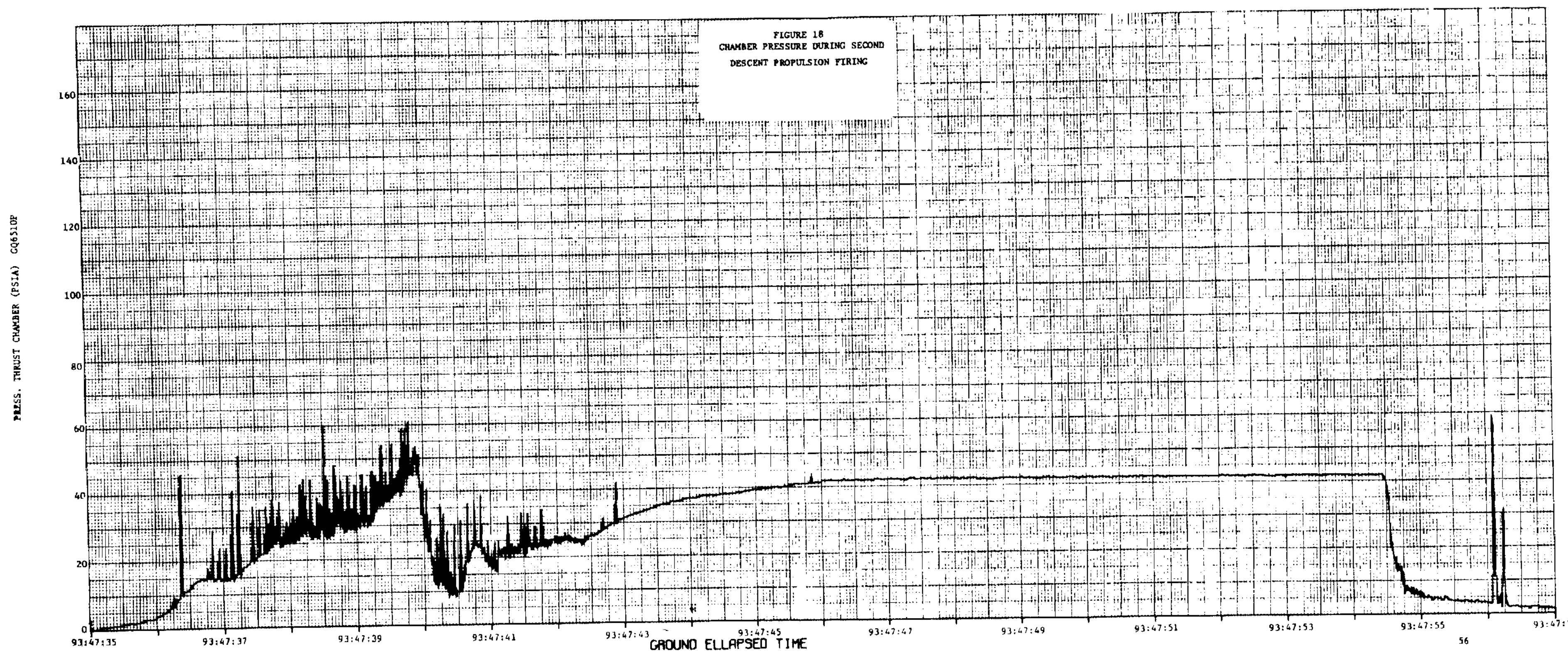


FIGURE 18
CHAMBER PRESSURE DURING SECOND
DESCENT PROPULSION FIRING



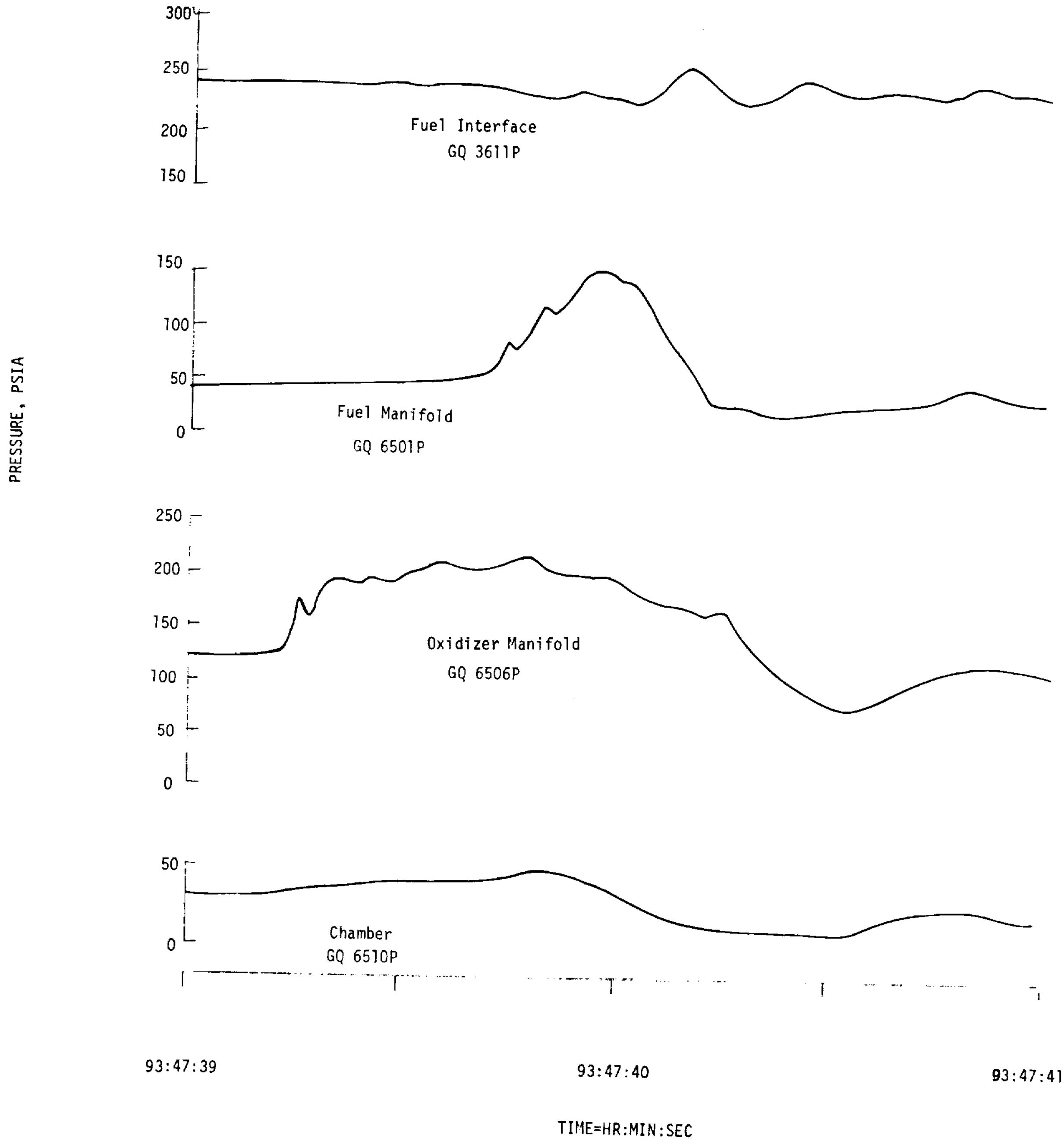


FIG. 19 Descent Propulsion System Pressures During Second Firing (Phasing Maneuver)

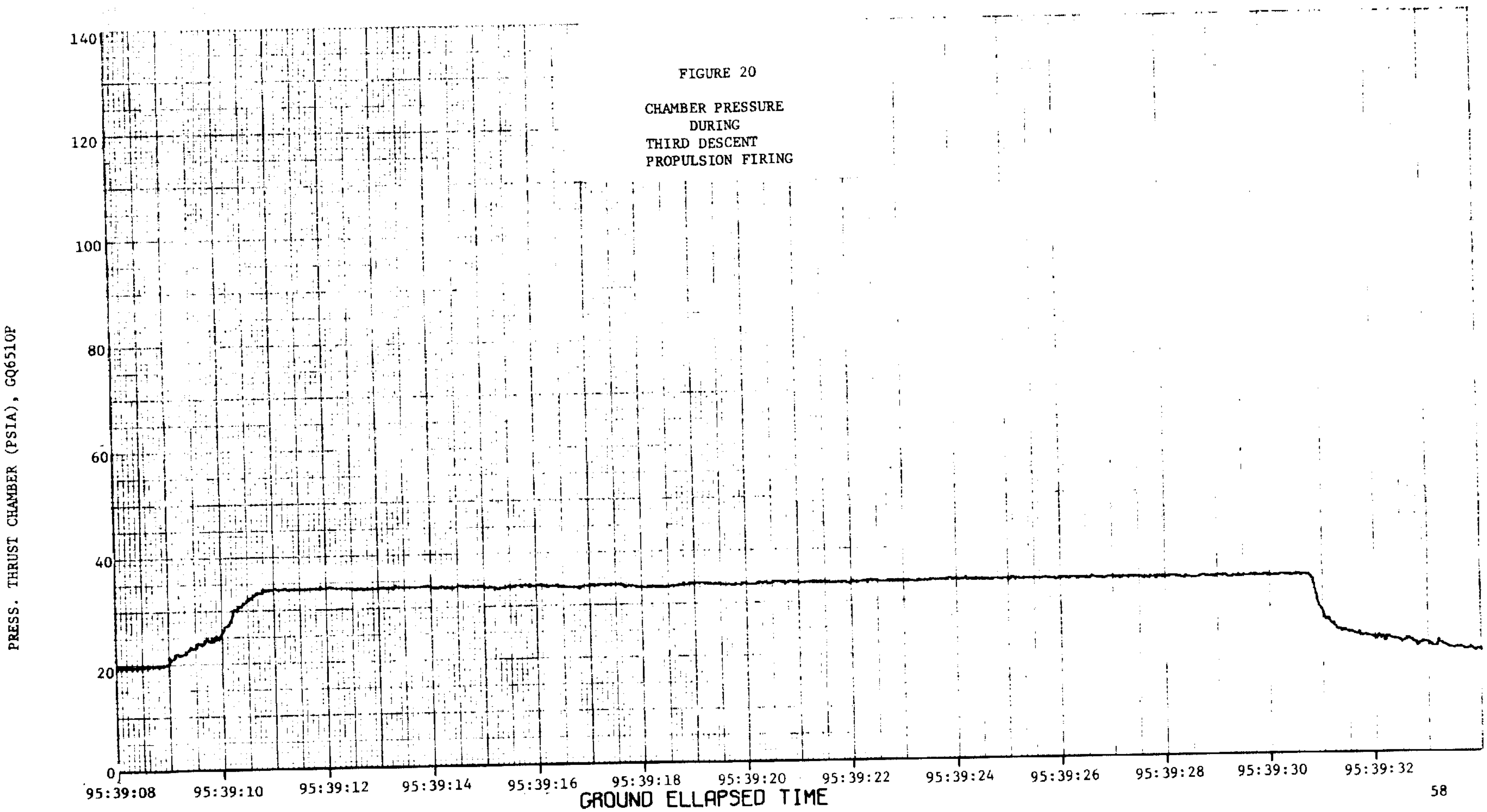


FIGURE 20

CHAMBER PRESSURE
DURING
THIRD DESCENT
PROPULSION FIRING

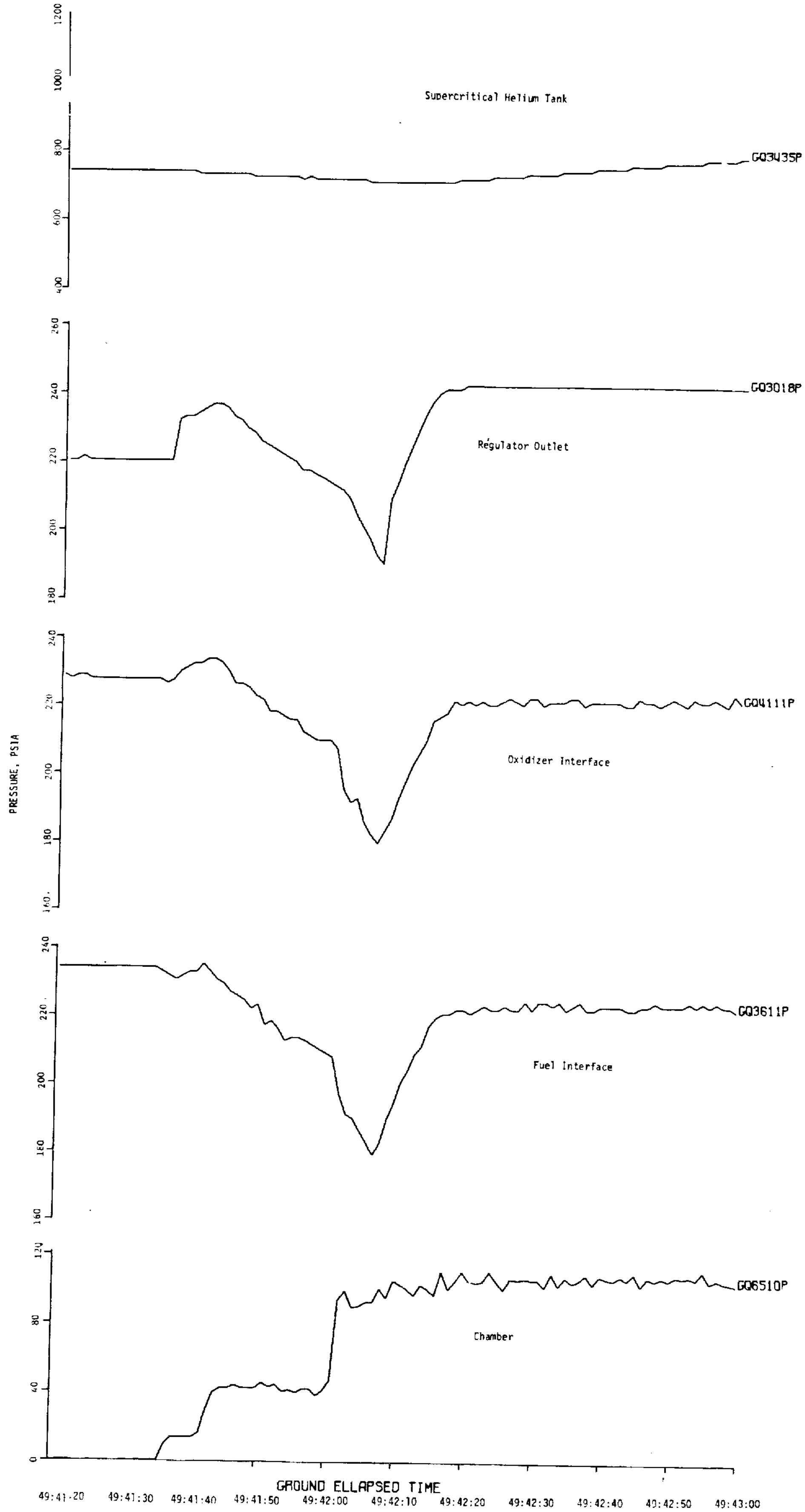


FIG. 21 Descent Propulsion System Pressures During Start of Docked Firing

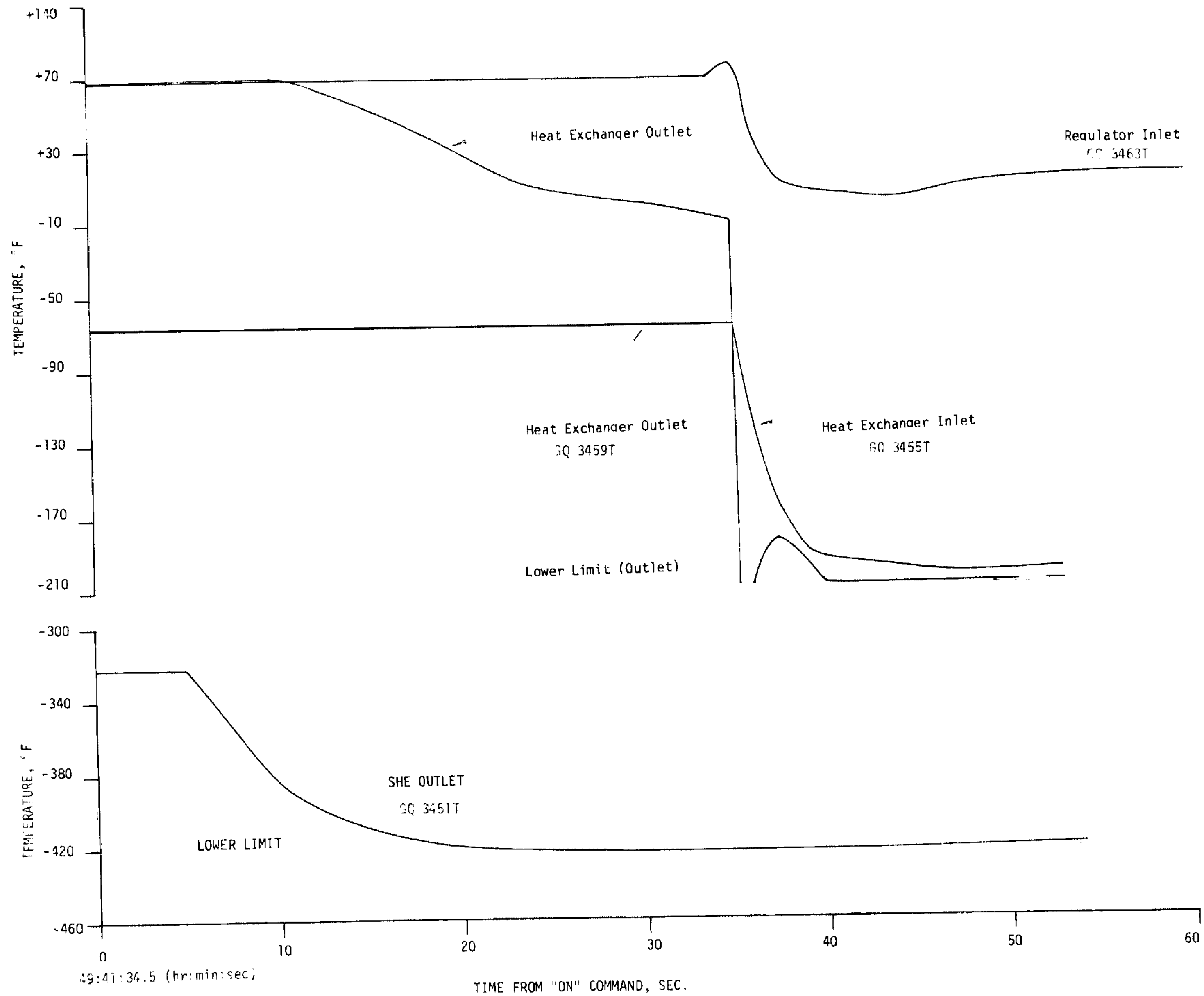


Figure 22 SUPERCRITICAL HELIUM SYSTEM TEMPERATURES DURING INITIAL PORTION OF DOCKED BURN